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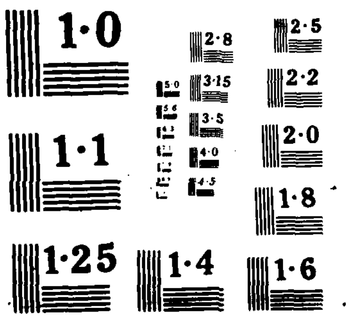
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AD-A172 865

FOREIGN TECHNOLOGY DIVISION

AIR FORCE SYSTEMS COMMAND

DESIGN OF AIRCRAFT
(Selected Chapters)

by

A.A. Badyagin, S.M. Yeger, et al.

FOREIGN TECHNOLOGY DIVISION

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PARTIALLY EDITED MACHINE TRANSLATION

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DESIGN OF AIRCRAFT (Selected Chapters)

By: A.A. Badyagin, S.M. Yeger, et al.

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In them occurs the state named "night blindness" - hemeralopia, - which, according to the current point of view, is a result of damage of the rod-shaped apparatus of the eye.

Page 51.

However, in recent years it has been shown that with the hereditary pigment degenerations in animals the biochemical changes are observed in all celluar elements of the retina.

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U. S. BOARD ON GEOGRAPHIC NAMES TRANSLITERATION SYSTEM

Block	Italic	Transliteration	Block	Italic	Transliteration
А а	*А а*	A, a	Р р	*Р р*	R, r
Б б	*Б б*	B, b	С с	*С с*	S, s
В в	*В в*	V, v	Т т	*Т т*	T, t
Г г	*Г г*	G, g	У у	*У у*	U, u
Д д	*Д д*	D, d	Ф ф	*Ф ф*	F, f
Е е	*Е е*	Ye, ye; E, e*	Х х	*Х х*	Kh, kh
Ж ж	*Ж ж*	Zh, zh	Ц ц	*Ц ц*	Ts, ts
З з	*З з*	Z, z	Ч ч	*Ч ч*	Ch, ch
И и	*И и*	I, i	Ш ш	*Ш ш*	Sh, sh
Й й	*Й й*	Y, y	Щ щ	*Щ щ*	Shch, shch
К к	*К к*	K, k	Ъ ъ	*Ъ ъ*	"
Л л	*Л л*	L, l	Ы ы	*Ы ы*	Y, y
М м	*М м*	M, m	Ь ь	*Ь ь*	'
Н н	*Н н*	N, n	Э э	*Э э*	E, e
О о	*О о*	O, o	Ю ю	*Ю ю*	Yu, yu
П п	*П п*	P, p	Я я	*Я я*	Ya, ya

*ye initially, after vowels, and after ъ, ь; e elsewhere.
When written as ё in Russian, transliterate as yё or ё.

RUSSIAN AND ENGLISH TRIGONOMETRIC FUNCTIONS

Russian	English	Russian	English	Russian	English
sin	sin	sh	sinh	arc sh	$\sinh^{-1}$
cos	cos	ch	cosh	arc ch	$\cosh^{-1}$
tg	tan	th	tanh	arc th	$\tanh^{-1}$
ctg	cot	cth	coth	arc cth	$\coth^{-1}$
sec	sec	sch	sech	arc sch	sech^{-1}
cosec	csc	csch	csch	arc csch	csch^{-1}

Russian	English
rot	curl
lg	log

GRAPHICS DISCLAIMER

All figures, graphics, tables, equations, etc. merged into this translation were extracted from the best quality copy available.

DESIGN OF AIRCRAFT.

A. A. Badyagin, S. M. Yeger, V. F. Mishin, F. I. Sklyanskiy, N. A. Fomin.

In the book general/common bases and methods of designing aircraft are presented, selection of diagram, power plant and fundamental parameters of aircraft is examined. The second edition includes the new materials: the methods of optimum design with the use of computer(s), the method of the gradients of takeoff weight for the evaluation of the designing solutions and conversion of weight characteristics, special feature of the design of aircraft with the shortened and vertical takeoff, the passenger and aerospace aircraft.

Sections, which relate taking into consideration of requirements of economy and to design of main aggregates/units, are considerably expanded and reworked.

Applications/appendices to book are supplemented by characteristics of aircraft engines, by standard combined weight and enumeration of standard electronic equipment.

Book is intended for students of aviation VUZ [- Institute of Higher Education] and can be useful for engineers of aircraft industry.

Page 3.

Preface.

In years, which passed from time of publication of textbook of N. A. Fomin "design of aircraft" (1961), significantly grew level of aviation science and technology. Appeared supersonic heavy aircraft, including passenger, usual became aircraft with the sweepback of wing variable in flight, are introduced VTOL aircraft.

Questions of design of aircraft within this time also obtained substantial development. Extensively are used the method of optimum design, systems approach, use of computer(s), etc.

Authors were aware in the fact that to write stable textbook according to design of aircraft is extremely difficult. Each decade in the aviation now whole epoch. Therefore in this book in comparison with the analogous previous textbooks considerable attention is given to the fundamental systematic questions, which are immune to so quick aging. At the same time the authors attempted to give material an information character, necessary for the diploma or pre-sketching design of aircraft.

Proposed textbook corresponds to program of course "Design of aircraft" for VUZ.

Textbook consists of two sections - general/common design of aircraft and design of its parts.

In first section of book three parts. In the first part general/common bases and methods of designing the aircraft are presented. Here the authors attempted in the intelligible form to show the evolution of the methods of general/common design, to present the bases of the procedure of optimum design during the identification of the parameters of aircraft. Is here given the procedure of the use of low increments in parameters and characteristics of aircraft, which makes it possible comparatively simple to solve the great circle of tasks.

In second part selection of diagram, power plant and fundamental parameters of aircraft is examined. Primary attention is here given to the design of jet aircraft. Separate of chapter they are dedicated to the special features of the general/common design of passenger aircraft, aircraft with shortened and vertical takeoff and landing, and also to the design of aerospace aircraft.

In third part of first section layout and position of center of gravity in aircraft is examined.

In second section of textbook procedure of determining dimensions and weight of wing, fuselage, tail assembly and landing gear is given.

Page 4.

The general/common bases of the design of the aircraft components and basis of the design of the system of its control here are examined.

Chapters I, III, IV, V, VI, XIV are written by A. A. Badyagin, introduction, Chapters II, VIII, XII, XIII - by N. A. Fomin, Chapter IX is written by S. M. Yeger, Chapters VII, XI - by V. F. Mishin, Chapters XVII, XIX - by F. I. Sklyanskiy. Chapters XVI and XVIII are written by A. A. Badyagin and V. F. Mishin, Chapter X is written by N. A. Fomin and N. K. Liseytsev, Chapter XV - N. A. Fomin and V. E. Rotin.

Appendices to textbook are written by A. A. Badyagin and V. F. Mishin.

All concrete/specific information on design of aircraft and selection of engine in period of preliminary development of aircraft is published in open Soviet and foreign press. The authors with the appreciation will accept the observations according to the book, which should be sent to an address: Moscow, B-66, 1st Basmannyy lane, 3, publishing house "Machine building".

Page 5.

The principal notations and contraction.

a - speed of sound, expenditures/consumptions per 1 ton- kilometer;

α - angle of attack of wing;

B - track of landing gear;

b - wing chord, the base of landing gear;

b_0 -root wing chord;

b_k – end wing chord;

C -cost/value;

$\bar{c}_0$ - wing chord ratio at the root;

$\bar{c}_k$ – relative thickness at the wing tip;

c_f – aerodynamic coefficient of friction;

c_m – coefficient of the aerodynamic pitching moment of wing profile;

c_{m0} – coefficient c_m when $c_y=0$

c_P – thrust coefficient;

c_e – the specific hourly consumption of fuel/propellant of TRD [turbojet engine];

c_e – the specific hourly consumption of fuel/propellant of TVD.

c_x – drag coefficient;

c_{x0} – drag coefficient when $c_y=0$

c_{xi} – the coefficient of induced drag;

$c_{xв}$ – coefficient of wave impedance;

c_{xp} – coefficient of profile drag;

c_y – lift coefficient;

c_y^{α} – derivative c_y on the angle of attack α;

D – diameter of fuselage;

δ - angle of deflection of any control;

E - modulus of normal elasticity of material;

F - area of the washed by flow surface;

f - coefficient of friction, the safety factor;

G - weight of aircraft;

G_0 - starting (takeoff) weight of aircraft;

G_{κ} – the structural weight;

$\bar{G} = \frac{G_{\kappa}}{G_0}$ – the over-all payload ratio of construction/design;

G_T – fuel load;

$\bar{G}_T = \frac{G_T}{G}$ – the over-all payload ratio of fuel/propellant;

g - acceleration of gravity; weight of 1 m^2 the surface of aggregate/unit;

γ - specific weight/gravity;

H - flight altitude;

χ - sweep angle of wing (on quarter-chord);

$\chi_{\pi.\kappa}$ – sweep angle (on the leading wing edge);

K - lift-drag ratio;

k - coefficient;

L - flying range, length;

l - wingspan;

λ - wing aspect ratio;

λ_i – elongation/aspect ratio of any aircraft component;

M - Mach number;

m - mass of flight vehicle; bypass ratio of TVRD;

m_x, m_y, m_z – the coefficients of the aerodynamic moment of aircraft (in the body coordinate system);

N - total power of engines.

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N_0 - starting power of engines (with V=0; H=0);

N_{01} - starting power of one engine;

$\overline{N_0}=N_0/G_0$ - relative starting power;

$n^p, n^э, n_y, n_x, n_z, n_A, n_E$ - load factors;

$n_{дв}$ - number of engines;

$n_{пас}$ - quantity of passengers;

P - total thrust of engines;

P_0 - the boost for launching of engines (with V=0; H=0);

P_{01} - the boost for launching of one engine;

$P_0=P_0/G_0$ - starting thrust-weight ratio;

$P_{уд}$ - specific thrust of power plant;

p - the specific wing load (p_0 - during the takeoff);

Q - hourly consumption of fuel/propellant;

q - velocity head, fuel consumption per kilometer;

ρ - mass air density; ρ_0 - on the surface of sea;

$\Delta=\rho/\rho_0$ - relative density of air;

R - radius of the Earth, a radius turn/bank and so forth;

S - wing area (with the subfuselage partly);

$\bar{S}$ - relative area of any aircraft component (referred to the wing area);

η - wing taper;

$\eta_в$ - efficiency of screw/propeller;

T, K - temperature in the degrees Kelvin;

T - the serviceable life of service; service life;

t - time, the temperature in the degrees Celsius;

θ - flight path angle to the horizon;

V - flight speed;

V_{1v} – orbital velocity;

V_v – the vertical velocity;

X - aerodynamic drag;

$x_{\cdot}$ – position center of gravity of aircraft from the leading edge of MAC;

x_F – the position of the focus of aircraft from the leading edge of MAC;

Y - aerodynamic lift.

Abbreviations.

VKS - aerospace of aircraft;

VPP - takeoff and landing strip;

GTD - gas turbine engine;

DTRD - turbofan engine;

ZhRD - liquid propellant rocket engine;

LPS - flight personnel;

PVRD - ramjet engine;

PD - piston engine;

PRD - solid propellant rocket engine;

SA - standard atmosphere;

MAC - the mean aerodynamic chord;

SPS - supersonic passenger aircraft;

TRD - turboprop engine;

TVRD - turbofan engine;

TRD - turbojet engine;

TRDF - turbojet reheat engine;

EVM - electronic computer;

UPS - boundary layer control;

SVVP - aircraft of vertical take-off/landing;

SUVP - aircraft of the shortened takeoff and landing.

Indices.

v - wave;

vzl. - takeoff (G_0 - takeoff weight);

v.o. - vertical tail assembly;

g - nacelle, load, throat;

g.o. - horizontal tail assembly;

dv - engine;

- interference;

k - construction/design.

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kach - rolling;

kom - commercial;

kr - wing;

kreys - cruising;

krit - critical (value);

m - midsection;

m.f. - midsection of fuselage;

0 - initial value of value (or with the start);

n - load;

n.v. - climb;

n.z. - navigational reserve;

ob. upr - equipment and control;

omyv - the washed surface;

op - tail assembly;

ost - stop;

otr - breakaway;

pas - passenger;

pl - gliding/planning;

p.n. - payload;

pos - landing;

pot - ceiling;

prerv - interrupted;

priv - given;

prob - landing run;

pust - empty;

rasch - calculated;

reys - scheduled speed;

razb - takeoff/run-up;

raskh - expended;

rl - control;

rch - knob/stick;

s.g - the jettisonable load (in flight);

sk - slip;

sluzh - official;

sn - equipment;

s.u. - power plant;

f - fuselage;

sh - landing gear;

ek - crew.

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Introduction.

By design of aircraft usually is understood process of development of technical materials (documentation), which determine its technical flight characteristics, diagram and construction/design of separate aggregates/units.

Designation/purpose, operating conditions and technical flight characteristics of projected/designed aircraft are determined by client and are designed in the form of special requirements.

Process of designing aircraft includes development of sketch and worker of projects. Work on the refinement of requirements for the aircraft and the possibility of their accomplishing, conducted to the beginning of the development of preliminary design, is called preliminary (pre-sketching) design.

Sketch design consists in development of fundamental characteristics of aircraft, its aerodynamic and design concepts, which make it possible to judge advisability of further design.

Into sketch design of aircraft it enters:

a) development of general views and layout sections/cuts;

b) reduced development of construction/design of most important parts (aggregates/units);

c) development of schematic diagrams, systems of equipment and control, and also power plant;

d) calculation of force of gravity (weight) and centering;

e) aerodynamic design, stability analysis and controllability;

f) approximate computation to strength of most important aircraft components.

Simultaneously with development of preliminary design aircraft scale model full size is constructed. For the examination of mock-up by client the board from different specialists is designated, including crew.

After conclusion of simulated board, which examines and confirms preliminary design and aircraft scale model, final fitting of construction with arrangement/position of control and equipment is accomplished/realized, external enclosures are more precisely formulated. Then conduct the more complete crews of aircraft on the strength, make and blow in the wind tunnels of model and according to the results of testings of model in wind tunnel they more precisely formulate aerodynamic design, stability analysis, spin and flutter. On the basis of the results of purgings the airplane design is more precisely formulated, the refined weight calculations are performed, in this case weight limits (greatest values of the weight of the structure of aircraft and its parts, the permissible from the considerations designs) are established/installed.

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Working design - this is completing process of development of technical documentation. Working project gives all necessary materials about technical flight data of future aircraft, about his strength and reliability; it contains equipment specifications and all necessary information for developing technology of production of aircraft. It must be noted that working the design of experimental aircraft usually concludes after that how the constructed sample of aircraft it underwent tests.

Into working design it enters:

a) development of assembly and detail drawings of construction/design of separate assemblies (parts) of aircraft;

b) development of general view drawings of assemblies of aircraft;

c) refinement of calculations for strength of all load-bearing elements;

d) refinement of calculations of structural weight;

e) conducting research and experimental works, connected with introduction of new constructions/designs, materials, etc.

Development of working design of contemporary aircraft extremely labor-consuming and complicated process, whose accomplishing under force only to large collective of qualified technical-engineering workers of different specialties.

Laboratories of design bureaus and scientific research institutes are occupied by experimental research works. The static and dynamic tests of construction/design for the strength, the service life, the reliability of separate assemblies and systems are the final stage of experimental research works usually. Hydraulic systems and other systems of equipment, system of control and recovery facilities aircrew in the necessary order undergo bench tests under the conditions, close to operating.

Weight immense space of knowledge, necessary for designing contemporary aircraft, was accumulated as a result of more than half century labor of scientific different countries - aerodynamicists, of material-strength engineers, metallurgists - and engineering practice of design, construction and production of aircraft.

Accumulation of knowledge and engineering experience contributed to improvement of aircraft, which was being accompanied by change in fundamental parameters and by improvement in its fundamental flight characteristics.

By fundamental aircraft performance usually are understood maximum speed of level flight V_{max}, ceiling $H_{пот}$, maximum vertical velocity $V_{y\,max}$ and maximum flying distance L_{max}. However, the fundamental parameters of aircraft include such parameters, whose change significantly is reflected in the characteristics of aircraft, in precisely: the takeoff weight of aircraft G_0, wing area S, the specific wing load $p_0=G_0/S$, horsepower loading G_0/N or thrust-weight ratio $\bar{P}_0=P_0/G_0$.

Tables 0.1 and 0.2 depict common picture of change in some of these values over years for heavy and light aircraft.

It is characteristic that one of fundamental aircraft performance (maximum speed of level flight) increased continuously from year to year.

Continuous increase in maximum speed of flight became possible as a result of decreasing aerodynamic drag of aircraft and decrease of load, which falls on 1 hp of installed power.

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Decrease of aerodynamic drag can be achieved/reached, first, by aerodynamic improvement of diagram and forms of aircraft, secondly, by decrease to known limit of load-bearing wing area (in the latter case specific wing load p_0 it is raised).

Most effective means of decrease of load on 1 hp was increase of installed power with decrease of value of its specific weight/gravity, i.e., relation of weight of installation and its power $G_{c.y.}/N_{max}$.

With increase in installed power usually grew/rose its weight, and also fuel load, necessary for achievement of prescribed/assigned range or duration of flight. At the same time grew/rose the weight of transportable by the aircraft of loads. All these factors, and increased requirements for the structural strength of aircraft, unavoidably led to an increase in the takeoff weight.

Decrease of aerodynamic drag of aircraft was achieved to a considerable degree as a result of aerodynamic improvement of wings, in particular wing profiles/airfoils.

Improvement of wings occurred simultaneously with development of airplane design, which in 30-35 years underwent considerable changes.

Used previously extremely widely diagram of biplane already since 1925 began to be displaced aerodynamic by more advanced schematic of cantilever monoplane with thick wing profile (of type of N. Ye. Zhukovskiy's profiles/airfoils), which had relative thickness $\bar{c}$=0.20-0.24.

Table 0.1. Change in the fundamental characteristics and parameters of heavy aircraft over the years.

(1) Год	(2) Наименование самолета	(3) G_0, кгс	S, м²	(4) p_0, кгс/м²	(5) V_{max}, км/ч	(6) N_{max}, л. с.	(7) G_0/N, кгс/л. с.	(3) P_0, кгс	(8) P_0/G_0, кгс/кгс	H, м
19[illegible]	«Гаккель-III» (Россия) (9)	42[illegible]	29	14,5	9[illegible]	35	12,0	—	—	1 000
1915	«Илья Муромец» (Россия) (10)	5 5[illegible]	1[illegible]	35,0	115	740	7,5	—	—	3 000
192[illegible]	К[illegible] (Италия) (11)	4 5[illegible]	[illegible]	47,[illegible]	[illegible]8[illegible]	600	7,5	—	—	4 000
19[illegible]	ТБ-3 (СССР) (11a)	[illegible]	23[illegible]	7[illegible]	24[illegible]	2000	9,0	—	—	4 000
194[illegible]	Б[illegible] (США) (12)	[illegible]	[illegible]	[illegible]	45[illegible]	4000	5,2	—	—	9 000
195[illegible]	B-29 (США) (12)	[illegible]	[illegible]	[illegible]	[illegible]	5000	5,9	—	—	10 5[illegible]
19[illegible]	[illegible] B-47 (США) (13)	[illegible]	[illegible]	[illegible]	9[illegible]	—	—	14 000	0,167	13 000
19[illegible]	[illegible] B-58 (США) (14)	[illegible]	[illegible]	4[illegible]	2000	—	—	36 000	0,5[illegible]	—
19[illegible]	XB-70 (США) (15)	25[illegible]	5[illegible]	427	[illegible]20[illegible]	—	—	82 500	0,[illegible]25	20 000

Key: (1). Year. (2). Designation of aircraft. (3). G_0 kg. (4). p_0 kgf/m². (5). km/h. (6). hp. (7). kg/hp. (8). kg/kg. (9). "Gakkel'-III" (Russia). (10). "Iliya Muromets" (Russia). (11). Caproni (Italy). (11a). TB-3 (USSR). (12). Boeing ... (USA). (13). Stratojet B-47 (USA). (14). Convair B-58 (USA). (15). XB-70 (USA).

Table 0.2. Change in the fundamental characteristics and parameters of light aircraft over the years.

(1) Гo	(2)	(3)	S	(4) p	(5) V	(6) N	(7) G/N	(3) P		H
1910	(8) [illegible]	[illegible]	29	[illegible]	9[illegible]	35	[illegible]	—	—	[illegible]
1915	(9) [illegible]	[illegible]	[illegible]	33.[illegible]	150	80	6.[illegible]	—	—	5[illegible]
1920	(10) [illegible] F-4 [illegible]	[illegible]	34.[illegible]	32.[illegible]	211	300	3,7	—	—	68[illegible]
1930	[illegible] (10a)	[illegible]	19.[illegible]	67.[illegible]	27[illegible]	525	2.5	—	—	8[illegible]
1940	(11) [illegible]	26[illegible]	[illegible]	159	570	[illegible]	2,3	—	—	[illegible]
1950	(12) [illegible]	48[illegible]	20.[illegible]	240	[illegible]	—	—	2000	0.435	[illegible]
1956	(13) [illegible] F-104 (США)	6[illegible]	17.[illegible]	40[illegible]	2500	—	—	9000	1.32	[illegible]

Key: (1). Year. (2). Designation of aircraft. (3). G_c kg. (4). p_c kgf/m². (5). km/h. (6). hp. (7). kg/hp. (8). "Gakkel'-III" (Russia). (9). Nieuport-21 (France). (10). Martinside F-4 (England). (10a). I-5 (USSR). (11). Messerschmitt 109 (Germany). (12). MiG-15 (USSR). (13). Lockheed F-104 (USA).

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Subsequently development of wings followed already path of gradual decrease of relative thickness of their profiles/airfoils (Fig. 0.1). Tendency toward decrease of $\bar{c}$ of wing profile is explained by the fact that with decrease of $\bar{c}$ shaped and wave wing drag decreases. Fig. 0.2 shows a change in the total coefficient of shaped and wave impedance [illegible] (when [illegible]) wing depending on $\bar{c}$. Especially sharply change $\bar{c}$ in the profile/airfoil affects the value of wave wing drag, which appears at the speeds, close to the speed of sound (curve M=1).

Transition/transfer from schematic of wing in the form of multi-stand biplane cell (Fig. 0.3a) to cantilever wing of first of thick (Fig. 0.3b), and then thin profile/airfoil (Fig. 0.3c) contributed to aerodynamic improvement of wing and decreased its overall height h. However, change in the overall height substantially affects weight of wing. With the decrease of structural height/altitude of wing h increase the forces, received with the bending by the elements of longitudinal wing skeleton, and, consequently, the weight of its structure increases. The application of thinner profiles/airfoils must lead to a considerable increase in the over-all payload ratio of wing construction, i.e., relation $G_{кр}\ G$

In actuality, as statistics shows, over-all payload ratio of cantilever wings not only did not increase, but, vice versa, it acquired tendency toward certain decrease.

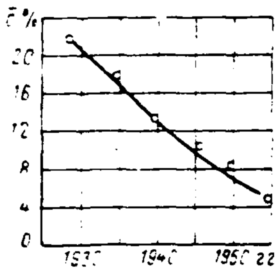

Fig. 0.1.

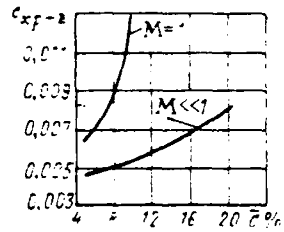

Fig. 0.2.

Fig. 0.1. Change in relative thickness $\bar{c}$ of monoplane freely load-bearing wing over years.

Fig. 0.2. Change in coefficient of shaped and wave impedance in dependence on thickness ratio with M<<1 and with M=1

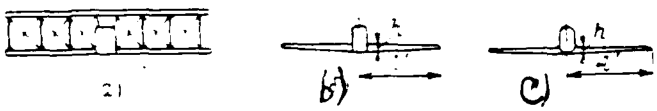

Fig. 0.3. Schematics of wings: a) multi-stand biplane; b) cantilever monoplane with thick wing (ratio of overall height to semispan h/l'=0.06); c) cantilever monoplane with thin wing (ratio of overall height to semispan h/l=0.035).

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This fact is explained by the following reasons:

1) by the increase in the specific wing load p_0, which occurred in the process of the development of aircraft construction;

2) by a gradual increase of the specific strength of the

materials, used in the construction of the aircraft;

3) by transition/transfer to the more rational structural load-bearing schematics of wings, and also by the improvement of the methods of the stress analyses of aircraft.

Increase in load on wing p_0 sufficiently significantly affects reduction in over all payload ratio of wing.

Fig. 0.4 shows curve $G_{кр}/G_{0}=f(p_{0})$, obtained as a result of processing statistical data on fighters with approximately equal values of fundamental parameters of wings, i.e., with approximately equal values of elongation/aspect ratio λ, thickness ratio $\bar{c}$ and load factor n_A.

Mechanical properties of materials, which were being used in aircraft construction, with years were improved. Its specific strength σ/γ [1] can be the characteristic (criterion), which defines in the first approximation, of the advantage of material in proportion by weight, as is known.

FOOTNOTE [1]. Specific strength with the extension characterizes the degree of the profitability of the application of material for the aircraft construction. Usually during the determination of the value of specific strength they accept dimensionality σ in kg/mm² and γ in G/cm², ENDFOOTNOTE. The greater the specific strength, the more advantageous the material for the application in the construction/design. The specific strength of aviation materials steadily is raised.

As a result of increase in specific wing load, increases in specific strength of materials, improvement of design concepts of wings, methods of calculations and tests for designers it was possible with considerable reduction in aerodynamic wing drag to preserve its over-all payload ratio approximately at one and the same level.

Decrease of aerodynamic wing drag in total resistance of aircraft was achieved not only as a result of transition/transfer to schematic of monoplane and decrease of thickness ratio of wing, but also as a result of continuous improvement of form of wing profiles/airfoils.

Large role in decrease of aerodynamic wing drag played different means of its mechanizations/lift-off devices (flaps, flaps, etc.), which made it possible to substantially increase specific wing load without considerable increase in landing speed of aircraft. One should note also that an increase in the specific wing load became possible because of the improvement of the landing equipment of the aircraft (application of wheel brakes, schematics of nose-wheel landing gear, etc.), with which was allowed/assumed certain increase in the landing speed.

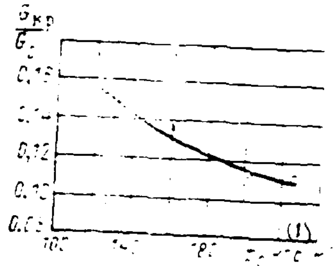

Fig. 0.4. Dependence of over-all payload ratio of wing construction G.:G on specific wing load p_0 of fighter-monoplanes, obtained as a result of processing statistical data.

Key (1). кГ/m^2.

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Fight for reducing aerodynamic drag of aircraft was conducted not only along line of reduction in wing drag, but also along line of reduction in drag of other aircraft components (fuselage, engine nacelles, landing gear, lamps/canopies, superstructures). Especially extensive work conducted on the decrease of drag of power plants. Won acceptance the cowlings of air-cooled engines with improved aerodynamics, ducted radiators for cooling the water and oil, arranged/located in the fuselage or in the wing.

Since 1931 designers began to extensively use retractable landing gear.

In proportion to perfection of technology of manufacture of aircraft improvement in the quality of finishing its external surfaces became possible. For the laminated enclosures this gave considerable effect in reduction in the frictional resistance.

Described above measures made it possible to considerably reduce drag coefficient of entire aircraft c_{x0} (Fig. 0.5), in spite of the fact that its value with increase in specific wing load p_0 must be to grow/rise [1].

FOOTNOTE [1]. Change p_0 differently affects the resistance of aircraft and drag coefficient c_x namely increase in p_0 always leads to increase/growth c_x whereas X with increase of p_0 at first decreases, and then with sufficiently high values of p_0 it begins to increase. ENDFOOTNOTE. From 1915 through 1950, i.e., in 35 years, the value of this coefficient was lowered from 0.033 to 0.015 (at the subsonic flight speeds).

In parallel with improvement of aircraft itself occurred improvement of aircraft engine. For an improvement in the aircraft performance it was necessary to increase the power of engine, to reduce its specific weight/gravity, to raise design altitude, i.e., to retain power up to highest possible heights/altitudes, and to decrease the specific fuel consumption.

Increase in power of piston engine is connected with gain in weight of power plant of aircraft [2].

FOOTNOTE [2]. Into the power plant of aircraft enter: the engines with the fastening and the screws/propellers, the throttle circuit, of lubrication system and fuel feed with the tanks, coolers, cowlings. ENDFOOTNOTE. If the power, which we want to obtain, is very great, then the weight of power plant so increases that the application of a piston engine proves to be unsuitable.

In order to avoid this, should be created lighter power plants (SU), than installation with piston engines (PD). As a result of investigations and experimental design works such SU was created on base of turbine jet engines (TRD).

Weight advantages of power plants with TRD in comparison with installations with PD become especially clear during comparison of specific weight of these installations, i.e., weight on 1 kG of engine thrust under flight conditions in earth/ground at different speeds (Table 0.3).

Large specific weight of power plants with PD and adverse characteristic of this type of engine on speed and height/altitude caused practical limit of maximum speed of "piston" aircraft approximately into 750 km/h.

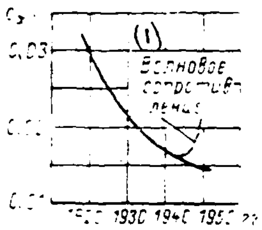

Fig. 0.5. Change in value of coefficient of drag without induced drag c_{x} $(c_y=0)$ over years.

Key: (1). Wave impedance.

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Considerably smaller specific weight of power plants from TRD (is at present equal to 0.25-0.35) and special feature of characteristics of thrust of these engines on speed and height/altitude they allowed aircraft not only to overcome sound barrier, but also to achieve high supersonic speeds.

Table 1.3. Specific weight of SU with TRD and PI in flight conditions near earth at speeds corresponding to different M Numbers.

V (1) км/ч	M	(2) Удельный вес СУ кгс/кгс тяги ПД	ТРД (1945 г.)	V (1) км/ч	M	(2) Удельный вес СУ кгс/кгс тяги ПД	ТРД (1945 г.)
0	0	0,9	0,5	720	0,59	3,0	[illegible]
360	0,3	1,5	0,5	900	0,74	3,9	[illegible]

Key: (1). km/h. (2). Specific weight/gravity of SU kG/kG thrust.

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Part I.

GENERAL DESIGN OF AIRCRAFT.

In practice of work of design bureaus for aircraft construction of many countries process of designing new sample of aircraft conditionally is subdivided into two stages: general/common design of aircraft and design of its individual parts (aggregates/units).

In general/common design determination of fundamental parameters and of aerodynamic configuration of aircraft, type of quantity and engine characteristics, layout and position of center of gravity in aircraft are included.

Section I.

THEORETICAL BASES AND METHODS OF THE GENERAL DESIGN OF AIRCRAFT.

Chapter I.

SHORT HISTORICAL COVERAGE OF THE DEVELOPMENT OF THE METHODS OF GENERAL DESIGN AND CRITERIA OF EVALUATION OF AIRCRAFT.

History of technology testifies that not one new machine, even if in it is used unusual principle, is not created at "empty place". This means that entire new in the technology carries some features of old and is based on previous experiment. For example, the first motor cars by the construction/design called to mind horse crews and bicycles, the first hydroturbines - water wheels of mills, etc. However, as far as the first aircraft are concerned, they to some degree were similar to the bats and some birds. Thus, K. Ader's aircraft "Aeolus" (1890) on the contours duplicated the silhouette of bat, and aircraft "Gakkel'-IX" (1912) called to mind in the plan/layout the predatory hovering bird.

Thus, first methods of general/common design of aircraft were based on laws of similarity and copying.

These methods sometimes were used also at more last stages of development of aviation, when according to one or the other reasons periods of design and construction of aircraft were too compressed.

At present methods of copying and similarity during creation of aircraft barely are used, since these methods exclude engineering search for best solutions on base of achievements of science and technology, and consequently, imply lag in this area of technology.

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Method of copying has still and that fundamental shortcoming, that at its base they lie/rest designation/purpose and operating condition of aircraft-original, which for aircraft-copy can be others.

It must be noted that similarity method and copying assumes invariability for aircraft-copy of some parameters of aircraft-original, for example elongation/aspect ratio, sweepback and wing chord ratio, although optimum values of these parameters depend on weight and sizes/dimensions of aircraft.

At present these methods are property of history.

Similarity methods and copying in 20-30's were substituted in general/common design by statistical method, known also by the name of method of prototypes. According to this method the parameters of the new sample of aircraft are obtained via their comparison with the statistical data of the constructed aircraft of analogous designation/purpose. Designer attempted to plan aircraft with better data than for existing.

According to this method fundamental parameters of aircraft should have been chosen, for example, with the help of numbers of Everling [36].

Some of creators and first propagandists of statistical method of general/common design were Soviet engineers P. M. Kreyson [13] and P. D. Samsonov [23].

If numbers of Everling of projected/designed aircraft and prototype aircraft did not coincide, then parameters were determined unsuccessfully and they changed taking into account of statistics or personal experience, then new calculation was done.

Weight of projected/designed aircraft during takeoff G_0 was determined in the first approximation, from relationship/ratio

$$G_0 = \frac{G_F}{k_1},$$

where G_F - weight of full load;

k_1 - statistical coefficient.

Knowing on statistics $p_0=G_0/S$, are found: area and span of wing $l=\sqrt{\lambda S}$. In this case wing aspect ratio λ also was determined according to the data of statistics or on the base of the experience of designers.

In second approximation/approach weight of aircraft during takeoff was determined in the form of sum of weights of its individual parts, i.e.

$$G_0 = k_2 G_{кр} + k_3 G_{ф} + k_4 G_{оп} + \cdots,$$

where k_2, k_3, k_4, ... - empirical coefficients;

$G_{кр}$; $G_{ф}$, $G_{оп}$ - weight of wing, fuselage, tail assembly respectively.

Statistical method of determining fundamental parameters of aircraft underwent development in works of A. L. Gimmel'farb [6]. The author of these works proposed the original method of the "modification of the aircraft of statistics" and conditions were indicated, at which any parameter of aircraft satisfies several requirements. For example, the specific wing load can be selected so as to satisfy the requirements maximum and landing speed. Subsequently statistical method was used for determining the takeoff weight of aircraft, different authors beginning to consider an increasing quantity of its components/terms (payload, the weight of power plant, equipment, crew, etc.).

Statistical method of layout assumes also assignment of technical flight data and parameters of projected/designed aircraft by extrapolation of possible within the next few years development of these data and parameters of constructed prototype so that up to moment/torque of issue series are new sample of aircraft it would not become obsolete. At the basis of statistical method the assumptions about a continuous, smooth (without the jumps) change in parameters and characteristics of prototype aircraft lie/rest.

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In the history of the development of aviation this period (to the jet aircraft) occurred and for it statistical method was justified. However, in the subsequent then period of the rapid intermittent development of aviation, statistical method in the general/common design of aircraft lost its value, since there is no prototype aircraft either completely or of them there are too few [1].

FOOTNOTE [1]. However, this does not mean that during the design the statistics should be rejected/thrown generally. ENDFOOTNOTE.

On the character of the development of aviation a great effect have two facts - increase in the complete cycle of the creation of the new sample of aircraft (from the beginning of design to the issue of a series) and a colossal increase in its cost/value.

If in 1940 this cycle amount on the average to four years on, then at present it grew of up to 6-10 years. It is obvious that the extrapolation of the flight characteristics of prototypes according to the data of statistics to 6-10 years - matter very unreliable.

Cost/value of program of creation of contemporary aircraft composes very large sum, which is evident from Table 1.1.

It is characteristic that each of aircraft indicated did not have predecessors (prototypes) and it is unique. Any state, such as rich it not was, cannot simultaneously create a large quantity of types of such aircraft, the cost/value of creation of which in the budget of the country becomes of ever of more ponderable [2].

FOOTNOTE [2]. For this reason some foreign states are forced to cooperate for the creation of aircraft (for example, SPS "Concorde", airbus A-300, etc.). ENDFOOTNOTE.

Creation of subsonic aircraft, although it costs less; however, not so that it would be economically profitably create simultaneously several samples of aircraft, which satisfy one and the same technical requirements.

Designs of aircraft now thoroughly are studied both with technical and from economic side and, as a rule, one project for realization is chosen.

Thus, statistics according to separate aircraft types becomes increasingly poorer and attempt unavoidably leads to use at it for creation of analogous sample to lag.

Method of optimum design of aircraft began to be developed from attempts at analytical solution of problems about selection of most advantageous parameters of aircraft taking into account contradictions between parameters and flight characteristics.

At basis of this method engineering search for parameters and performances (object, process), which in the best way satisfy selected evaluation criterion, lies/rests.

Table 1.1.

Название самолета (1)	Расчетное число M полета (2)	Взлетный вес самолета в тс (3)	Стоимость программы создания самолета в млрд. долл. (4)	Источник (5)
SR-71 (США)(6)	3	45	~1	Interavia, № 5545, 5546, 1964 г.(7)
Б-70 «Валькирия»(8) (США)(6)	3	250	1,5	Flight, № 2885, 2886, 1964 г.(7)
TSR-2 (Англия)(9)	2,2	45	~1	Aviation Week, 4.XI. 1963 г.(7)
(10) Боинг 2707 (США) (проект)	2,8—3,0	306	1,45	Interavia, № 6245, 1967 г.(7)
(11) «Конкорд» (Франция — Англия)	2,0—2,2	159	1,6	Interavia, № 6082, 1966 г.(7)

Key: (1). Name of aircraft. (2). Calculated flight Mach number. (3). Takeoff weight of aircraft in t. (4). Cost/value of program of creation of aircraft in billions of dollars. (5). Source. (6). USA. (7). g. (8). Valkyrie. (9). England. (10). Boeing 2707 (USA) (project). (11). "Concorde" (France-England).

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By evaluation criterion certain objective function, which expresses fundamental quality of object of design (for example, aircraft), is implied.

First Soviet work, dedicated to optimization of parameters of aircraft, is work of Soviet engineer N. N. Fadeyev "research of suitable sizes/dimensions of aircraft" (1939). Then, in 1940-1942 appeared the analogous work of other authors. From this time special sharpness and importance gained a question about the criteria of evaluation of aircraft, since the optimum design without the criteria is impossible.

In work indicated N. N. Fadeyev for the first time indicated output from numerous contradictions with selection of characteristics and parameters of aircraft. He proposed all those contradicting to each other the parameters and characteristics to join into a certain generalized criterion.

Later to the same conclusion about need for single complex criterion arrived S. I. Zonshayn, V. F. Bolkhovitinov et al.

If for statistical methods intuition of designer played significant role, since with the help of intuition compromise solutions more or less correctly were located, then with the advent of method of optimum design role of intuition of designer substantially becomes narrow, being inferior place for scientifically substantiated solution (which, is understood, on it excludes creative beginning in research, setting and methods of solution of problems of design).

Development of method of optimum design was actually introduction in prospecting work of designer of dialectical method, at base of which lies/rests idea about interdependence and interconditionality of all aircraft quality/fineness ratios.

In 1946 Soviet designer V. F. Bolkhovitinov for the first time showed [4] that if we in equation of weight balance of aircraft express weight components through flight characteristics and other parameters, then in modified equation, named equation of existence, it is possible not only to see connections/communications between different properties of aircraft, but also to judge possibility of realization of these properties at the level of development of aviation technology in this period. This means that one or the other aircraft quality/fineness ratio can be achieved/reached only with the specific weight perfection of aircraft components.

With the help of equation of existence it is possible to answer also question, is possible realization of flight vehicle generally.

At the end of fortieth years during design of flight vehicles it arose and idea of comprehensive investigations began to be developed. The essence of this idea consisted in finding of the analytical connection/communication between the qualitatively different properties of aircraft, for example between the weight and the cost/value.

To wide use of computer(s) in design of aircraft method of optimization of parameters was developed in essence along line of research of new and improvement of known relations between the parameters of aircraft. The optimization of a large quantity of parameters during calculations by hand represented and presents great calculating difficulty, especially during the simultaneous optimization of several parameters. Output, true, was in the approximate, consecutive optimization of the series/row of the parameters. However, the authentic solution of this task of optimization became possible only with the application of electronic digital calculators (1960-1961), which can after several hours calculate of thousands and tens of thousands of versions of the combinations of parameters and characteristics of aircraft, they can automatically develop the results of calculations and give out them in the form (in the form of tables, graphs or drawings) convenient for the designer.

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Because of computer(s) are opened/disclosed new qualitative and quantitative possibilities of solution of different problems at design of any machine and, in particular, during identification of parameters of aircraft, since identification of parameters is reduced to comparison and evaluation of diverse variants. Computer makes it possible by one-two orders to increase a number of investigated alternative designs and to select the most adequate/approaching version. The total number of possible combinations of the parameters is expressed, as is known, by the product

$$n = n_a n_b n_c \ldots n_i,$$

where $n_a, n_b, n_c, \ldots, n_i$ - numbers of versions (unknown) parameters being investigated or layouts a, b, c, ..., i. If, for example, there are investigated four versions of values for each of six unknown parameters, then the total number of combinations is equal $4^6=4096$.

Within compressed periods, which usually are let for sketch design of aircraft, to investigate several thousand combinations of parameters and layouts in order to select best version, it is virtually possible only with the help of computer(s).

Thus, computer allows to considerably larger degree, than during manual calculation, to carry out principle of optimum design.

Application of computer(s) gives possibility to turn from averagings and simplifications to calculations, generally accepted by manual calculations, gives possibility to take into account not only effect of fundamental, but also secondary dependences of criterion of evaluation of aircraft on aerodynamic, weight and other characteristics.

As has already been indicated, question about criterion or criteria of evaluation of projected/designed aircraft is one of most important questions, since on it not only numerical value of parameters and characteristics depends, but also fate of projected/designed or constructed aircraft generally.

Unsuccessfully selected criteria can lead to completely incorrect evaluation of aircraft.

Cases, when unsuccessfully formulated criteria retarded development of good aircraft and encouraged development of aircraft, rejected subsequently by practice, are known. For example, the Soviet attack aircraft Il-2, constructed in 1938, was considered as the some military specialists unsuccessful, on the basis of its flight characteristics. The application of the Il-2 in the patriotic war showed the salient military characteristics of this aircraft. Unsuccessful proved to be not aircraft, but attempt to consider it only according to the flight-performance data.

Are known opposite examples.

Hence follows important conclusion: criterion of evaluation of aircraft must consider not only flight characteristics, but also conditions for application (operation).

First aircraft were evaluated according to separate characteristics. Aircraft was considered best, if it had the highest speed, the best load ratio with the equal range, etc in comparison with the prototypes. The comparison of aircraft according to numbers of Everling (to numbers of speed, range, height/altitude) was spread. Subsequently by Soviet scientist V. S. Pyshnov were proposed analogous Everling numbers of coefficients [21], which make it possible judge the value of payload, the structural weight, the aerodynamic cleanness of aircraft, about that, did strive designer for obtaining of maximum speed, maximum load capacity or he made compromise decision. V. S. Pyshnov's coefficients, in addition to this, they gave the possibility to consider the fuel consumption per ton- kilometer transportable load, and also the quality of propeller.

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Numbers of B. Everling as V. S. Pyshnov's coefficients, were used during design of aircraft on base of statistical method of selection of his fundamental parameters and characteristics and in their time played progressive role.

It must be noted that numbers of Everling and coefficients of Pyshnov gave possibility to evaluate and to be congruent/equate aircraft according to different criteria (besides economic), but they did not give possibility to objectively consider aircraft as a whole.

Actually, let us assume that there is some quantity of evaluation criteria. If criteria equally considered aircraft quality/fineness ratios, it would be easy to do a selection. But what is to be done, if number or coefficients are the most varied? Some criteria give high evaluation, others - low.

In these cases were recommended these methods:

- the making of certain average compromise decision, based on experiment and intuition;

- to use evaluation according to system of points, placing criteria in order of their importance.

First method is purely subjectivist and does not lead compulsorily to best solution. As early as 1952 N. N. Fadeyev noted that not average/mean, but "the extreme solutions are often best" [29].

Second method claims to certain objectivity; however, at base entire its the same subjectivism, on which is based determination of importance of criteria and appropriation with it of "specific weight/gravity". This method becomes completely ineffective in the cases, when fundamental requirements for the aircraft are implemented, and rest, more numerous, are not implemented. It is unclear, what is more important - to satisfy the first, fundamental requirement, or all rest?

It is obvious that all these difficulties can be surmounted, if we take single, sufficiently general/common criterion of evaluation, where will be reflected everything interesting designer and operating personnel of characteristic and parameters of aircraft. This general/common, synthetic, criterion must not be artificial. It is necessary that it would express primary task, for which is created flight vehicle. For the military aircraft - these are combat efficiency or the degree (completeness) of the fulfillment of the combat mission, for commercial airplane - economical transportation of passengers and loads on the prescribed/assigned level of comfort and with fulfilling of all requirements of safety and regularity of flights.

Soon, after appearance of numbers of Everling, which were, actually, technical indices, were done propositions about economic evaluation of transport aircraft. The first economic criterion was proposed K. Rokk in the work "air traffic from the point of view of the economy" (1929).

Known at present criterion (prime cost of ton-kilometer) introduced into Soviet aviation literature P. N. Tolmazov in work "fundamental questions of operation of airlines", ONTI, 1934. Then Ye. A. Ovrutskiy [16] dedicated monograph to the development of the criterion of the economic evaluation of transport aircraft.

As criterion of comparative evaluation of commercial airplanes Ye. A. Ovrutskiy accepts prime cost ton-kilometer a as value, which directly reflects necessary expenses for creation and operation of aircraft [see formula (1.2)].

This criterion is sufficiently general/common and at the same time not not too complicated, completely available during design of aircraft.

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The prime cost of ton-kilometer considers the weight characteristics of aircraft and its parts, characteristic of engines (thrust, the specific fuel consumption, cost/value, service life), the aerodynamic characteristics of aircraft (through the fuel load), flight conditions (V, H, L), cost/value and service life of the construction of the aircraft, expenditures for the design of aircraft, to the content of crew and stewards, the value of fuel/propellant, the airport expenditures/consumptions, average annual load factor. Takeoff and landing characteristics for the airfield of any prescribed/assigned class are considered indirectly, since thrust-weight ratio and wing loading per one square meter, on which depends the engine thrust and the weight of aircraft, affect the level of the prime cost of ton-kilometer. The requirements of safety and comfort of passengers are indirectly considered also, since the engine failure is one of the conditions for the selecting of thrust-weight ratio. Into fuel load emergency navigational reserve enters, is provided for a gain in weight in the fuselage construction due to the organization of the ejection hatches and other recovery facilities. On the prescribed/assigned level of the comfort of passengers (versions: the first class, tourist, economic) depend either the sizes/dimensions of fuselage (structural weight, aerodynamics) and the weight of standard equipment with the given number of passengers, or the weight of payload and weight of general equipment with the constant sizes/dimensions of fuselage.

Comparative estimate of commercial airplanes at cost of transportation is official and abroad. In the USA and other countries there are standard methods of determining the straight/direct operational expenditures/consumptions for a comparative evaluation of transport aircraft.

In comparison with prime cost of ton-kilometer, criterion, proposed by D. L. Tomashevich [25], is more general economic criterion. It takes the form:

$$\eta = \frac{P}{B}.$$

where P - value, which determines the public usefulness or the purposeful output/efficiency of flight vehicle;

B - expenditure for the manufacture of flight vehicle and the maintenance of its efficiency in the period of operation.

D. L. Tomashevich's criterion is applicable to any flight vehicles or articles generally. However, in the application/appendix to commercial airplanes the criterion in question represents not that another as the value, reciprocal of the complete prime cost of ton-kilometer 1/a.

In the case of comparison of aircraft with constant values of flying range, cruising speed, payload, cost/value and resource/lifetime of parts economic criteria transform themselves into simpler criterion - takeoff weight of aircraft. The best version corresponds to the minimum of the takeoff weight of aircraft, other conditions being equal. In many instances the weight criterion G_0 substantially simplifies the solution of problems by the optimization of the parameters and structural-design solutions.

For military aircraft in series/row of countries are developed/processed general/common criteria, which consider probability (degree) of execution of assignment, which depends on characteristics of combat load and equipment, aircraft performance, and also characteristics of ground-based air safety equipment of flight [4, 25].

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That presented makes it possible to do following conclusions:

- methods of general/common design of aircraft were developed along path of analysis from simple to more complicated - from methods of copying and similarity to statistical method and then - to method of optimum layout;

- criterion of evaluation of aircraft times be brought down along path of synthesis from particular to general/common - from evaluation according to separate signs/criteria to evaluation of aircraft as a whole.

In conclusion let us present procedure of calculation of contemporary criterion of evaluation of commercial airplanes - reduced expenditures, at basis of which prime cost of ton-kilometer lies/rests, but are considered also capital investments.

Reduced expenditures, expressed relative to one aircraft, are equal to

[illegible] to kopecks/ton-kilometer. [illegible]

Here a - prime cost of transportation in kopecks/ton-kilometer;

$a_{прив}$ - given investments in kopecks/ton-kilometer.

Prime cost of transportation is designed from formula

$$a=\frac{A}{K_{ом}G_{ком}V_{рейс}} \quad \text{kopecks/ton-kilometer,} \quad (1.2)$$

where A - expenditures for operation of aircraft for one flying hour in kopecks/h;

$G_{ком}$ - payload in t, which corresponds to this flying range;

$V_{рейс}$ - scheduled speed of aircraft in km/h;

$K_{ом}$ - coefficient of payload, which considers average annual incomplete load of aircraft due to seasonality of transportation.

Values A are given in Table 1.2.

Scheduled speed of aircraft or flight speed according to schedule considers time losses to following stages of flight: starting/launching and engine warm-up, taxiing to takeoff and landing strip (runways) before takeoff and after landing, takeoff and climb, maneuvering in air after takeoff and before landing, reduction/descent and landing. For the supersonic aircraft the time to the dispersal/acceleration of aircraft to the cruising supersonic flight speed and to the braking additionally is considered. Scheduled speed is determined from the following formula:

$$V_{рейс} = \frac{L}{\frac{L - L_{вп}}{V_{крейс}} - t_{вп} - \Delta t_{м}} = \frac{L V_{крейс}}{L - L_{вп} - (t_{вп} - \Delta t_{м}) V_{крейс}} . \qquad 1.3$$

where L - distance between the airports of takeoff and landing in km;

$t_{вп}$ - time, spent on the takeoff, the climb, reduction/descent and landing, in h;

$L_{вп}$ - horizontal projection of the path, passed by aircraft for time $t_{вп}$ in km;

$V_{крейс}$ - cruising flight speed in km/h;

$\Delta t_{м}$ - time, spent on starting/launching and engine warm-up, for the taxiing and the maneuvering after takeoff and before the landing in h.

Time for climb and reduction/descent, to dispersal/acceleration of aircraft to cruising speed both braking and corresponding to this time horizontal projection of path of aircraft are taken from aerodynamic design.

$\Delta t_{м}$ is taken as equal to:

for aircraft with TRD - 10 min (0.167 h),

for aircraft with TVD - 15 min (0.25 h).

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For rough calculations it is possible to accept that scheduled speed is equal to:

$$V_{\text{комм}} = \frac{L}{L/V_{\text{кр}} - \Delta t} = \frac{L V_{\text{кр.ср}}}{L - \Delta t V_{\text{кр.ср}}} \quad \text{km/h,} \qquad (1.4)$$

where Δt=20 min for aircraft with TRD (0.33 h); - 35 min for main-line aircraft with TVD (0.58 h); - 10 min for light multipurpose aircraft with PD and TVD (0.167 h).

Maximum payload is determined depending on quantity of passenger places and capacity/capacitance of luggage and cargo compartments on aircraft

$$G = 90 n_{\text{па}} - 290\left(v_{\text{а}} - \frac{20 n_{\text{па}}}{120}\right) \quad \text{kg,} \qquad (1.5)$$

where $n_{\text{па}}$ - number of passenger places;

90 - average weight of passenger (75 kg) and personal luggage of passenger (15 kg);

290 - average proportion of mail and load in kG/m³;

$v_{\text{а}}$ - space of luggage and cargo compartments in m³;

120 - average specific weight/gravity of luggage of passengers in kG/m³.

Passenger aircraft can transport maximum payload to specific, so-called calculated, range, which depends on maximum takeoff weight and on fuel reserve on aircraft. With the prescribed/assigned maximum takeoff weight the distance flight, larger than calculated, is made with the reduced payload as a result of the appropriate increase in the fuel reserve.

Standard graph, which shows dependence of value of payload of passenger aircraft from flying range, it is shown in Fig. 1.1.

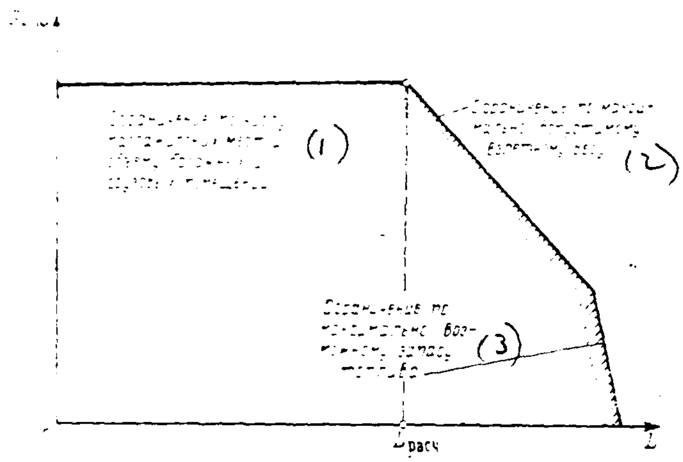

Fig. 1.1. Standard dependence of value of payload of passenger aircraft from flying range ($L_{расч}$ corresponds to minimum of prime cost of transportation).

Key: (1). Limitation according to a number of passenger places and by the space of luggage and cargo compartments. (2). Limitation on maximum permissible takeoff weight. (3). Limitation on maximally possible fuel reserve.

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Expenditures for operation of aircraft for one flying hour A consist of: expenditures for shock absorption of aircraft $A_{a.c}$ and engines $A_{a.д}$ maintenance cost and aircraft maintenance $A_{т.о.с}$ and engines $A_{т.о.д}$ cost/value of expendable fuel/propellant $A_т$ wage of crew with countings $A_{зп}$ and indirect (airport) expenditures/consumptions $B_{ап}$ which consider expenditures for content of airports and different administrative-technical services of airline. Thus

$$A = A_{a.c} + A_{a.д} + A_{т.о.с} + A_{т.о.д} + A_т + A_{зп} + B_{ап}$$ kopecks/h.

Expenditures for shock absorption of aircraft are designed from formula

$$A_{a} = K_1 C \frac{1 - K_{p\,c}\left(\frac{T_c}{t_c} - 1\right)}{T_c} \text{ Kopecks/h} \quad (1.7)$$

where K_1=1.05 - coefficient, which considers non-production flying time (agings/trainings, instruction, flight around, etc.);

C_c - cost/value of aircraft without engines in kopecks:

$$C_c = K_{cep\,c} K_V G_{пуст} (3340 - 0{,}077 G_{пуст} - 1.05 \cdot 10^{-4} G_{пуст}^{1.5}). \quad (1.8)$$

Here $G_{пуст}$ in kg; $K_{cep\,c}$ and K_V - coefficients, which consider seriality and rated speed of the flight of the projected/designed aircraft:

$$K_{cep\,c} = \left(\frac{35 \cdot 10^5}{G_{пуст} \Sigma n_c}\right)^{0.4}; \quad (1.9)$$

$$K_V = \frac{1}{2}\left(1 + \frac{V_{крейс}}{800}\right), \quad (1.10)$$

where Σn_c - quantity of aircraft in a series;

$V_{крейс}$ - cruising speed in km/h.

In formula (1.7) coefficient $K_{p\,c}$ -ratio of cost/value of one major overhaul to first cost of aircraft - can be calculated by formula

$$K_{p\,c} = 0.11 \frac{[illegible]}{C_c}; \quad (1.11)$$

where T_c - amortization or complete service life of aircraft in h;

t_c - service life of aircraft between major overhauls in h. For the main-line aircraft in average/mean T_c=30000 h, t_c=500 h. For the aircraft of local airline $T_{c}=25000$ h, $t_{c}=5000$ h.

By analogy (1.7) are designed expenditures for shock absorption of engines

$$A_{дв}=K_{2}n_{дв}C_{дв}\frac{1-K_{...}\left(\frac{T_{дв}}{t_{пр}}-1\right)}{T_{дв}} \text{ kopecks/h,} \quad (1.12)$$

where K_2=1.07 - coefficient, which considers non-production flying time;

$n_{дв}$ -number of engines, established/installed on aircraft;

$C_{дв}$ - cost/value of one engine in kopecks.

For TRD it is possible to accept

$$C_{дв}=K_{cx}K_{cep.дв}P_{0}(3400-10\sqrt{P_{0}}) \text{ kopecks,} \quad (1.13)$$

where P_0 - takeoff thrust of one engine in kg;

K_{cx} and $K_{cep.дв}$ - coefficients, which consider type (diagram) of engine and seriality; $K_{cx}=1$ - for TRD; $K_{cx}=1.15$ - for DTRD with M<1; $K_{cx}=1.5$ - for engines of supersonic passenger aircraft;

$$K_{cep.дв}=\left(\frac{1500}{\Sigma n_{дв}}\right)^{0.?} \quad (1.14)$$

Here $\Sigma n_{дв}$ - quantity of engines in a series.

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Cost/value of TVD on the average is equal to

$$C_{p} = 300 N_{0} \text{ to kopecks,} \quad (1.15)$$

where N_0 - takeoff power of one engine in hp. In formula (1.12) coefficient K_{1} - the ratio of the cost/value of one engine overhaul to its first cost.

For TRD and DTRD

$$K_{1} = 0.15 - 4.15 \cdot 10^{-5} [1 - 0.2 (\frac{T_{3P}}{t_{3P}} - 1)] T_{3P}. \quad (1.16)$$

where T_3 - amortization or complete service life of engine in h;

t_3 - service life of engine between major overhauls in h.

For calculations is received $T_{3P} = 6000$ h and $t_{3P} = 3000$ h, i.e., it is considered that in its service life engine is repaired one time.

For TVD it is possible to accept $K_{1} \approx 0.6$

Maintenance cost and aircraft maintenance are determined from formula

$$A_{c} = K_3 K_4 (0.39 - 0.121 \cdot 10^{-5} G_{r}) G_{r} \text{ kopecks/h,} \quad (1.17)$$

where K_3=0.5 - coefficient, which considers method of maintenance, used in this airline. During the introduction of the progressive forms of maintenance this coefficient can be reduced to 0.35- 0.40;

K_4=1 - for subsonic aircraft with TRD and DTRD;

K_4=1.13 - for the aircraft with TVD;

K_4=2.0 - for supersonic passenger aircraft (M≥1.8).

Maintenance cost and maintenance of engines are determined from formula

$$A_{т.о.д}=\frac{16K_2K_5n_{дв}\sqrt{P_0}}{1-7\cdot10^{-5}\cdot T_{дв}}\text{ KopecKs/h, }\quad(1.18)$$

where K_2=1.07 - as in formula (1.12);

K_5=1 - for TRD and DTRD of subsonic aircraft;

K_5=1.5 - for engines SPS, and also for TVD. For the aircraft with TVD in (1.18) instead of P_0 one should take N_0 in hp.

Expenditures for wage of crew $A_{зп}$ are designed, on the basis of number of members of flight personnel (pilots, navigators, flight engineers and radio operators) - $n_{л.п.с}$ and number of stewards - $n_{бп}$:

$$A_{зп}=\bar{C}_{л.п.с}n_{л.п.с}+\bar{C}_{бп}n_{бп}\text{ kopecks/h.}\quad(1.19)$$

Here $\bar{C}_{л.п.с}$ and $\bar{C}_{бп}$ - mean hour wage of flight personnel and stewards. Values $\bar{C}_{л.п.с}$ and $\bar{C}_{бп}$ with average/mean flying time of crew 550 h per annum are given in Table 1.2.

Table 1.2.

Назначение самолета (1)	$K_{к.м}$	$\bar{C}_{з.п.с}$ коп/ч (2)	$\bar{C}_{б.п}$ коп/ч (2)	b	K_8	K_9
(3) Магистральный дозвуковой	0,56	1100	400	1,30	2700	0,42
(4) Магистральный сверхзвуковой	0,65	2000	750	1,30	2700	0,42
(5) Для местных авиалиний	0,65	1100	400	1,27	2600	0,53
(6) Легкий многоцелевой ($n_{пас} \leqslant 6$)	0,75	800	—	1,23	2000	0,61

Key: (1). Designation/purpose of aircraft. (2). kopecks/h. (3). Main-line subsonic. (4). Main-line supersonic. (5). For feeder lines. (6). Light multipurpose.

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Cost of fuel/propellant expendable in flight is determined from formula:

$$A_{т} = 5.1\,K_{6}\frac{G_{т.расх}V_{рейс}}{L_{расч}}$$ kopecks/h, (1.20)

where $G_{т.расх}$ - fuel/propellant expendable in flight in kg, it undertakes from aerodynamic design taking into account further expenditure/consumption time Δt [see formula (1.4)];

5.1 - cost/value 1 kg of fuel/propellant (taking into account non-production expenditures for testing of engines, on training and check flights) in kop/kgs;

K_6=1.0 - for subsonic aircraft with TRD and DTRD;

K_6=1.03 - for aircraft with TVD;

K_6=1.06 - for supersonic passenger aircraft.

Indirect (airport) expenditures/consumptions are taken as equal to:

$$B_{ar} = K_{\cdot}(0.45 - 7 \cdot 10^{-4}\sqrt{G_0})\,G_0 \quad \text{kopecks/h,} \quad (1.21)$$

where G_0 - in kg; K_7=1.0 - for subsonic aircraft; K_7=1.5 - for supersonic.

Given investments, entering in (1.1), are determined from formula

$$[illegible] = \frac{E\,[illegible]C\,[illegible]}{[illegible]} \frac{[illegible]C\,[illegible]n\,b)}{[illegible]} \quad \text{kopecks/t·km.} \quad (1.22)$$

Here E - standard coefficient of the efficiency of investments; E=0.121/year ($E=1/T_{H}$, where T_{H} - standard payback period the investments in the years);

C and $C_{дв}$ - cost/value of aircraft and engine [formula (1.8), (1.13), (1.15)] V kopecks;

K, G, V -see (1.2);

b - ratio of a number of engines, intended for operating the aircraft taking into account replacements, to a number of engines, established/installed on the aircraft. For $T_{дв} \geq 4000$ h values b are given in Table 1.2;

$B_{год}$ - time in the air to the aircraft per annum,

$$B_{r} = K_{\cdot}\frac{L_{a[illegible]}}{L_{pacч} - K_{\cdot}V_{peйc}} \quad \text{h/year,} \quad (1.23)$$

where L - calculated flying range for the projected/designed aircraft (see Fig. 1.1) in km;

$V_{\text{рейс}}$ - scheduled speed in km/h;

K_{ξ}, K_{ξ} - coefficients (see Table 1.2).

Chapter II.

Fundamental and relative parameters of aircraft, equation of over-all payload ratios. Effect of the most important parameters of aircraft on its flight characteristics.

Most important task of designing aircraft - determination of its fundamental parameters: takeoff weight G_0; wing area S; thrust P_0 or power N_0 of power plant, required for obtaining prescribed/assigned flight characteristics.

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These parameters serve as initial values for developing entire design of aircraft, and their correct selection causes optimum technical indices and flight-performance data of aircraft.

In certain cases at design to more conveniently use relative parameters: by specific load on wing area $p_0=\frac{G_0}{S}$; thrust-weight ratio $\bar{P}=\frac{P}{G}$; thrust, in reference to wing area $\frac{P}{S}$; load on power $\frac{G}{N}$.

§ 1. Equation of the over-all payload ratios of aircraft.

Complete (takeoff) weight of aircraft is composed of several those differing in its special features of parts

$$G_0=G+G+G+G\ldots$$

where G - weight of structure of aircraft;

G - weight of power plant;

G - fuel load;

G - weight of equipment, equipment, crew and cargos.

Value G depends on series/row of parameters of aircraft and its parts, mainly, from specific wing load p_0, of wing aspect ratio λ, of coefficient of calculated g-force n_p of weight of aircraft, etc. Value G depends on the weight per horsepower, on the thrust level of aircraft, on the weight of tanks and so forth; G_T - from specific consumption of fuel, range of aircraft, from its cruising speed, from the weight of aircraft, etc. Value G directly with parameters and characteristics of aircraft and with its weight is not connected and is determined depending on the aircraft type and its designation/purpose.

If we divide both parts of given equality to G_0, then will be obtained equation

$$1=\bar{G}+\bar{G}+\bar{G}_T+\bar{G}$$

called equation of over-all payload ratios of aircraft or equation of weight balance of aircraft. In this case the relationships/ratios

$$\bar{G}=\frac{G}{G_0};\ \bar{G}_{c.y}=\frac{G_{c.y}}{G_0};\ \bar{G}_T=\frac{G_T}{G_0};\ \bar{G}=\frac{G}{G_0}$$

are the respectively over-all payload ratios of the construction/design of aircraft, power plant, reserve of fuel, equipment, crew and loads. The equation of over-all payload ratios, as it will be shown below, plays considerable role in the disclosure/expansion of the dependence between the parameters and the aircraft performance.

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§ 2. Dependence of fundamental aircraft performance on the separate parameters.

Maximum speed.

Maximum speed of level jet flight (when it is not limited to heating construction/design, by flight safety, etc.) at height/altitude H can be determined according to formula

$$V_{max}=\sqrt{\frac{16P}{c_x S\Delta}}=\sqrt{\frac{16r\bar{P}}{c_x\Delta}}, \qquad (2.1)$$

where P - full thrust of power plant of aircraft at given height/altitude H at speed V_{max};

S - wing area;

c_x - coefficient of drag of aircraft at speed V_{max};

$\bar{P}=P/G_0$ - available thrust- weight ratio of aircraft in flight at height/altitude H with speed of V_{max};

$\Delta=\frac{\rho}{\rho_0}$ - relative density of air at height/altitude H.

Thrust of jet engines (TRD and PVRD) depends on speed and flight altitude. With an increase in the velocity of flight V and corresponding to it number M(V=aM, where a - speed of sound), the thrust of engines P at speeds M≤0.5 somewhat is reduced, and then, depending on the parameters of engine, it grows, after which on large Mach numbers sufficiently sharply it falls. In the initial stage of the design of aircraft it is convenient a change in the thrust to judge according to the graph of coefficient $\xi=\frac{P}{P_0}$, where P_0 - starting (with the work on the spot) thrust (for TRD - with M=0). The exemplary/approximate graph of change ξ on M for H≥11000 m is given in Fig. 2.1. Thus, the thrust of all engines at any speed with H=0 can be expressed thus:

$$P_V=\xi P_0$$

A change in the thrust of TRD on height/altitude H occurs in accordance with the formula

$$P_H=P_0\Delta^{\epsilon},$$

where the exponent ϵ at heights H≤11000 m somewhat less than one $(\epsilon\approx0.85-0.9)$, and at heights H≥11000 m it is equal to one. In subsequent presentation the tasks, connected with the flight at heights/altitudes H>11000, will be examined for the large part. Consequently, expression for the engine thrust at any speed and at any height/altitude H>11000 can be written thus:

$$P=m\xi P_0\Delta,$$

where m - numerical coefficient, determined by the change-in-thrust pattern at heights H<11000 m, and ξ=f(M) will have identical character for the engines with similar parameters.

After dividing both parts of this equality to G_0 and after taking

ϵ=0.85 and m=1.2, we will obtain

$$\text{(1) для } H < 11000 \text{ (2) м } \quad \bar{P} = \xi\Delta^{0,8\epsilon}\bar{P}_0; \qquad (2.2)$$

$$\text{(1) для } H \geqslant 11000 \text{ (2) м } \quad \bar{P} = 1{,}2\xi\Delta\bar{P}_0. \qquad (2.2')$$

Key: (1). for. (2). m.

where $\bar{P}_0 = \frac{P_0}{G_0}$.

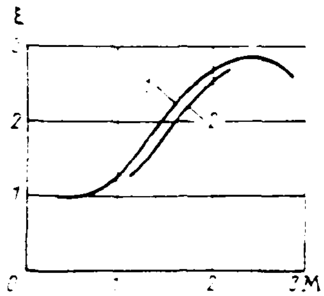

Fig. 2.1. Dependence ξ (ratio of thrust level in flight P to thrust with work on the spot P_0) on speed for TRD (degree of increase of pressure in compressor $\pi_k=6$ temperature of gases before turbine $T_3=1200°$, height H≥11000 m): 1 - without taking into account entry loss; 2 - taking into account entry loss.

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After substituting into formula (2.1) obtained expression (2.2'), let us be able to write for heights H≥11000 m

$$V_{max}=15.7\sqrt{\frac{p_0\bar{P}_0\xi}{c_x}}\ \text{km/h}, \quad (2.3)$$

or

$$M_{max}=0.0148\sqrt{\frac{p_0\bar{P}_0\xi}{c_x}}; \quad (2.3')$$

for the heights/altitudes H<11000 m

$$V_{max}=14.4\sqrt{\frac{p_0\bar{P}_0\xi}{c_x\Delta^{0,15}}}\ \text{km/h}, \quad (2.4)$$

$$M_{max}=\frac{4}{a}\sqrt{\frac{p_0\bar{P}_0\xi}{c_x\Delta^{0,15}}}, \quad (2.4')$$

where a - speed of sound at heights H<11000 m. In this case

$$c_x = c_{x0} + c_{xi} = c_{x0} + D_0 c_y^2,$$

where.

$$D_0 = \frac{c_{xi}}{c_y^2}.$$

Ceiling.

For aircraft with TRD static ceiling $H_{пот}$ is determined by value of relative density on ceiling, which can be obtained according to formula

$$\Delta_{пот} = \frac{1.60 \sqrt{D_0 c_{x0}}}{\xi \bar{P}_0}. \quad (2.5)$$

of that escaping/ensuing from obvious equality $P = \frac{G_0}{K_{max}}$. By using (2.2') and by assuming/setting $K_{max} = \left(\frac{c_y}{c_x}\right)_{max} = 0.5/\sqrt{D_0 c_{x0}}$, we will obtain formula (2.5).

Vertical rate of climb.

Maximum vertical velocity approximately can be expressed so [5]:

$$V_{y\,max} \approx \frac{2}{3} \frac{P}{G} \sqrt{\frac{2P}{3\rho c_{x0} S}}.$$

Assuming/setting for $H=0$ $\rho = \frac{1}{8}$; $P/G = \xi \bar{P}_0$; $P/S = p_0 \bar{P}_0 \xi$, we will obtain

$$V_{y\,max} = 1.53 \sqrt{\frac{p_0 (\bar{P}_0 \xi)^3}{c_{x_0}}}, \quad (2.6)$$

or for the engine with afterburner $V_{y\,max} = 1.53 \sqrt{\frac{p_0 (\bar{P}_{0ф} \xi)^3}{c_{x_0}}}$, where $\bar{P}_{0ф}$ - thrust-weight ratio with afterburner.

Maximum range at speed $V_{крейс}$

The work, produced by the engine thrust in flight of aircraft up to distance of L, can be expressed thus:

$$L' = 1.53\ P_{ср} L,$$

where $P_{ср}$ - average thrust on path of L.

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The same work, expressed through mechanical heat equivalent of burned down fuel/propellant, is determined by expression

$$L' = 427 G_т H_т \eta_0.$$

where $H_т$ - fuel heating value;

$G_т$ - weight of burned down fuel/propellant;

η_0 - complete efficiency of power plant.

After equating right sides of obtained expressions, let us find

$$L = \frac{427\eta_0 H_т G_т}{P_{ср}}, \qquad (2.7)$$

whence, in particular, is visible dependence of range L on calorific value $H_т$. Formula (2.7), however, cannot serve as base for the examination of the effect of different parameters of aircraft on its range and therefore let us turn to the detailed analysis of following formula [13] for the aircraft with the jet engines (TRD and PVRD):

$$L = 3.6 \int_{[illegible]} \frac{[illegible]}{c} \frac{[illegible]}{c_x} V \frac{dG}{G}\ км, \qquad (2.8)$$

where V - speed in the m/s;

c_i - specific hourly consumption of fuel/propellant in kg per 1 kg of thrust;

G_1 - weight of aircraft at the end of the path; if we consider that in the path the weight changes only due to the fuel consumption, then

$$G_1 = G_0 - G_T$$

During design of aircraft maximum flying distance is of great interest. It is obvious that for achievement of maximum range it is necessary that in the prescribed/assigned ratio G_1/G_0 product $\frac{1}{c_i}\frac{c_y}{c_x}V$ in formula (2.8) would have maximum value. Let us explain, with what of condition X expression $\frac{1}{c_i}\frac{c_y}{c_x}V$ has a maximum. Let us consider for this the factors, which affect the entering it values.

From theory of TRD it is known that specific fuel consumption depends on flight speed V, heights/altitudes H (or Δ) and degree of throttling/choking engine (or number of revolutions n). Graphically these dependences are represented in Fig. 2.2. As can be seen from this graph, c_p with increase V increases whereas (see Fig. 2.2a), with an increase in the height/altitude to H=11000 m - is gradually decreased, at the heights/altitudes H>11000 m c_p it remains the constant (see Fig. 2.2b).

During throttling/choking of engine, i.e., with gear down noticeably it changes, being at first reduced, in this case minimum value $n.c_p$ it is reached at small degree of throttling/choking in c_p so-called cruise setting of engine, and then rapidly increasing (see Fig. 2.2c).

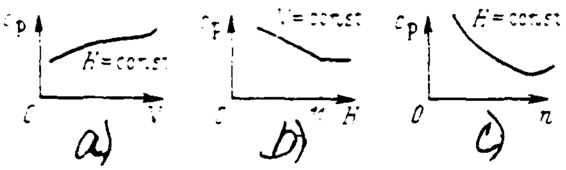

Fig. 2.2. Dependence c_p on speed V (a), height/altitude H (b) and number of revolutions of TRD n (c) for single-flow engines.

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Value of relation c_y/c_x depends on flight conditions, i.e., from coefficient c_y on which is accomplished flight. The expression, which determines c_y can be obtained from the relationships/ratios

$$\frac{c_y}{c_x}=\frac{G}{P};\quad \frac{G}{P}=\frac{1}{\bar{P}}.$$

Hence $c_y=\frac{c_x}{\bar{P}}$.

Replacing in this equality c_x by its expression through c_{x0} and $D_0c_y^2$ we will obtain

$$c_y=\frac{c_{x0}+D_0c_y^2}{\bar{P}},$$

whence

$$c_y=\frac{\bar{P}}{2D_0}-\sqrt{\frac{\bar{P}^2}{4D_0^2}-\frac{c_{x0}}{D_0}}. \qquad (2.8')$$

After substituting expressions for (2.2) and (2.2') into formula (2.8'), we will obtain

$$c_y=\frac{\bar{P}_0\Delta^{0.85}}{2D_0}-\sqrt{\frac{\bar{P}_0^2\Delta^{1.7}}{4D_0^2}-\frac{c_{x0}}{D_0}} \quad (H\leqslant 11000\ \text{м});$$

$$c_y=\frac{0.6\bar{P}_0\Delta}{D_0}-\sqrt{\frac{0.36\bar{P}_0^2\Delta^2}{D_0^2}-\frac{c_{x0}}{D_0}} \quad (H>11000\ \text{м}).$$

These formulas show that c_y depends on $\bar{P}_0$, heights/altitudes H (or Δ) and on the flight speed V, on which in turn depend ξ_j, c_{x0} and at supersonic speeds and D_0.

Fig. 2.3 depicts curve, which gives representation about character of change c_y in level flight according to heights/altitudes in cruise setting of TRD. Obviously, c_y on ceiling $H_{\text{пот}}$ will achieve greatest value $c_{y\,\text{доп}} = c_{y\,\text{опт}}$, its value when K_{max}. With the decrease of height/altitude H coefficient c_y is reduced and respectively is reduced quality c_y/c_x.

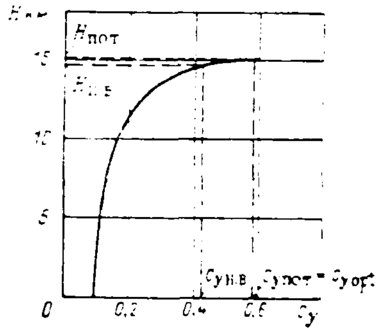

Fig. 2.3.

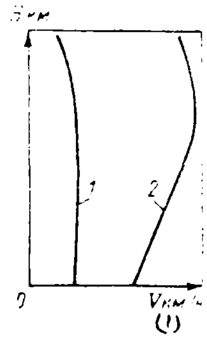

Fig. 2.4.

Fig. 2.3. Change c_y in level flight with completely open choke/throttle of TRD in dependence on height/altitude H ($H_{нв}$ - most advantageous height/altitude).

Fig. 2.4. Character of change in maximum speed in dependence depending on height/altitude H: 1 - transonic aircraft; 2 - supersonic aircraft.

Key: (1). km/h.

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Speed of level flight V_{max} at the given height/altitude for the aircraft with TRD, as noted, can be expressed by formulas (2.3) and (2.4).

After using values c_x of aircraft according to blasting or approximate computation and by curves of dependence of coefficient ξ on speed, it is possible to graphically determine speeds of level flight at different heights/altitudes.

In this case for series/row of selected heights/altitudes in the range from H=0 to H=11000 m, being assigned by values of speeds $V_{заз}$, we determine by corresponding value c_x and ξ and, substituting in formula (2.4), we find values $V_{нст}$, in terms of which we plot a curve in coordinates $V_{заз}$ and $V_{нст}$. Then, using formula (2.3), let us perform the same operation for heights H>11000 m. For each selected height/altitude we will obtain graph in coordinates $V_{заз}$ and $V_{нст}$. After leading on each graph from the origin of coordinates ray/beam at an angle of 45°, we will obtain in the point of intersection of ray/beam and curve the V value of speeds V at the selected heights/altitudes H.

Fig. 2.4 gives typical curves of change along height/altitude of maximum speed for subsonic (1) and supersonic (2) aircraft. For both aircraft types an increase in the speed along the heights/altitudes to H=11000 m is characteristic.

On the basis of aforesaid higher for aircraft with TRD are valid following conclusions:

a) specific consumption of fuel c_T with increase in altitude to H=11000 m is reduced also at heights H>11000 m it becomes constant (see Fig. 2.2);

b) coefficient c_y, which corresponds to level flight, with an increase in height/altitude H it increases to $c_{y\,эт}$ (see Fig. 2.3), reaching this value on the static ceiling. Together with c_y with an increase in altitude increases aircraft quality/fineness ratio c_y/c_x, which on the ceiling takes maximum value;

c) the speed of the level flight V with an increase in altitude to H=11000 m increases.

Taking into account given conclusions/derivations, it is possible to confirm that product $\frac{1}{c_p}\frac{c_y}{c_x}V$, and consequently, and flying range they reach maximum value at heights H>11000 m.

Assuming that specific consumption of fuel c_p for heights H≤11000 m can be expressed thus:

$$c_p \cong \psi\Delta^k c_{p0},$$

where $\psi=1.05+0.1M+0.05M^2$ is considered approximately effect of flight Mach number in the range from 0.8 to 3.0 on specific hourly consumption;

Δ - relative density of air;

$$k=0.12;$$

c_{p0} - starting specific expenditure/consumption (M=0; H=0), we can for H≥11000 m write

$$c_p=0.863\psi c_{p0},$$

i.e. at heights H>11000 m specific expenditure/consumption does not depend on height/altitude.

Formula (2.8) now can be obtained in the following form:

$$L=4.17\int_{G_1}^{G_0}\frac{1}{\psi c_{p0}}\frac{c_y}{c_x}V\frac{dG}{G}\ \text{км}. \qquad (2.9)$$

Keeping in mind, that $\frac{c_y}{c_x}V$ on a change in weight G does not depend, let us integrate this expression, assuming that $G_1=G_0-G_T$, $V=\text{const}$:

$$L=4.17\frac{1}{\psi c_{p0}}\frac{c_y}{c_x}V\ln\frac{G_0}{G_0-G_T},$$

or

$$L = 4.17 \frac{1}{c_{p_0}} \frac{c_y}{c_x} V \ln \frac{1}{1 - \bar{G}_T}. \quad (2.9')$$

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Maximum value of distance L_{max} will occur with maximum value $\left(\frac{c_y}{c_x} V\right)_{max}$.

Substituting for V expression (2.3), we will obtain

$$\frac{c_y}{c_x} V = 15.7 \frac{c_y}{c_x^{1.5}} \sqrt{p_0 \bar{P}_0 \xi},$$

whence it follows that, since value $p_0 \bar{P}_0$ for this aircraft can be accepted constant/invariable $\left(\frac{c_y}{c_x} V\right)_{max}$, it will occur when $\left(\frac{c_y}{c_x^{1.5}} \xi^{0.5}\right)_{max}$. Consequently, for obtaining maximum range L_{max} aircraft with TRD must fly at conditions, with which $\left(\frac{c_y}{c_x^{1.5}} \xi^{0.5}\right)_{max}$.

Let us determine c_y, which corresponds to this regime. obviously, maximum range will be obtained when $\left(\frac{c_x^{1.5}}{c_y \xi^{0.5}}\right)_{min}$. After raising this fraction into degree of 2/3 and using an analytical expression of the polar

$$c_x = c_{x_0} + D_0 c_y^2, \quad (2.10)$$

we will obtain

$$\left(\frac{c_x^{1.5}}{c_y \xi^{0.5}}\right)^{2/3} = \frac{c_{x_0}}{c_y^{2/3} \xi^{1/3}} + \frac{c_y^{4/3}}{\xi^{1/3}} D_0. \quad (2.11)$$

For the subsonic aircraft the coefficient ξ on the speed changes little and it is possible in the first approximation, to take $\xi \approx 1$, and $D_0 = \frac{1}{\pi \lambda_{eff}} = \text{const}$ and c_x depends only on c_y. We differentiate expression

(2.11) on c_{y}, let us make equal to zero and we will obtain most advantageous value $c_{y\,НВ}$ for regime L_{max}:

$$c_{y\,НВ}=1{,}252\sqrt{\lambda_{эф}c_{x_0}}.$$

Value $c_{y\,opt}$, corresponding to maximum quality, as is known, will be equal to

$$c_{y\,opt}=\sqrt{\pi\lambda_{эф}c_{x_0}}=1{,}773\sqrt{\lambda_{эф}c_{x_0}}. \qquad (2.12)$$

Consequently,

$$c_{y\,НВ}=\frac{1{,}252c_{y\,opt}}{1{,}773}=0{,}71c_{y\,opt}.$$

Thus, flight to maximum range of subsonic aircraft must be produced under the conditions, during which $c_{y\,НВ}=0{,}71\ c_{y\,opt}$, and height/altitude $H_{НВ}$ is somewhat less $H_{пот}$ (see Fig. 2.3) with work of engine in cruise setting.

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Substituting (2.12) into theoretical expression of polar (2.10), we will obtain value $c_{x\,НВ}$ for regime of maximum range after simple conversions

$$c_{x\,НВ}=1{,}5c_{x_0}.$$

In transonic zone of speeds (M≈0.8-1.3) coefficient ξ noticeably changes. Furthermore, in this range c_{x0} and D_0 depending on speed endure considerable changes; therefore for the transonic aircraft the determination of value c_y of the regime of maximum range, i.e., regime $\left(\frac{c_y}{c_x}V\right)_{max}$, it can be carried out in a following simple graphic manner. Having a polar of aircraft on the blasting or according to the data of the approximate computations for the speeds in the range M≈0.8-1.3, we plot a curve $c_x=c_x(M)$ for the series/row of heights H>11000 m (Fig.25).

Then to the same graph it is applied curve $c_P = c_P(M)$ for H>11000 m, calculated by the formula

$$c_P = \frac{P}{Sq} = \frac{19.2 p_0 \bar{P}_0 \xi}{(aM)^2}. \qquad (2.13)$$

Curve $c_P = c_P(M)$ we construct for this value of thrust-weight ratio $\bar{P}_0$ so that it would intersect curves $c_x = c_x(M)$ in transonic speed range. We find values ξ from the graph $\xi = \xi(M)$, then the points of intersection of curves $c_x = c_x(M)$ and $c_P = c_P(M)$ will correspond to flight speed M at different heights/altitudes H. Now, knowing M, it is possible for each of the heights/altitudes undertaken to find values c_y from the formula

$$c_y = \frac{16 p_0}{\Delta (aM)^2}$$

and, consequently, to determine qualities $\frac{c_y}{c_x}$ corresponding to them.

Further let us compute for series/row of values c_y of value $\frac{c_y}{c_x}V$ and let us construct curve $\frac{c_y}{c_x}V$ depending on c_y (Fig. 2.6) [1].

FOOTNOTE [1]. Fig. 2.6 as an example gives curve for the hypothetical transonic aircraft. ENDFOOTNOTE.

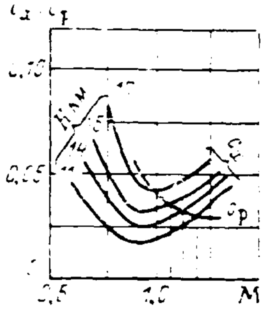

Fig. 2.5.

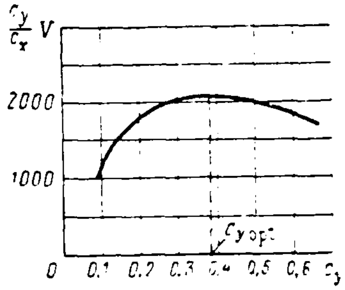

Fig. 2.6.

Fig. 2.5. Determination of dependence c_x on M from different heights/altitudes for prescribed/assigned engine (auxiliary construction for obtaining dependence, given in Fig. 2.6).

Fig. 2.6. Dependence $\frac{c_y}{c_x}V$ on c_y for determination c_y corresponding to most advantageous flight conditions to maximum range.

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With the help of this curve let us determine coefficient $c_{y\,HB}$ to which it corresponds

$$\left(\frac{c_y}{c_x}V\right)_{max}$$

and in this case let us ascertain that $c_{y\,HB}=c_{y\,opt}$.

Consequently, in transonic zone regime of maximum quality will be most advantageous regime for flight (with H≠const), i.e., for achievement L_{max} flight must occur with

$$c_y=c_{y\,opt}.$$

If flight of transonic aircraft occurs at speed, at which sufficiently noticeable matched impedance appears, then regime when $c_{y\,opt}$ with completely open choke/throttle due to sharp increase in matched impedance becomes unfavorable and maximum range is obtained during certain throttling/choking of engines, greater than in cruise setting.

For supersonic zone $c_{y\,\text{НВ}}$ it is possible to find by graphic method so, as this is described above.

Results of determination from this method $c_{y\,\text{НВ}}$ show that maximum $\left(\frac{c_y}{c_x}V\right)$ occurs with

$$c_{y\,\text{НВ}} = 0.73 c_{y\,\text{opt}},$$

i.e. it is possible to approximately consider that regime of maximum range for supersonic aircraft is analogous to regime of maximum range of subsonic aircraft. Allowing/assuming a certain unessential inaccuracy, it is possible during determination L_{max} of supersonic aircraft in the first approximation, to accept

$$c_{y\,\text{НВ}} = 0.71 c_{y\,\text{opt}}.$$

Thus, we will assume that regime of maximum range is characterized both for subsonic and for supersonic aircraft with TRD by coefficients $c_{y\,\text{НВ}} = 0.71\ c_{y\,\text{opt}}$ and $c_{x\,\text{НВ}} = 1.5\ c_{x0}$. To these values $c_{y\,\text{НВ}}$ and $c_{x\,\text{НВ}}$ corresponds value $\left(\frac{c_y}{c_x}\right)_{\text{НВ}}$:

$$\left(\frac{c_y}{c_x}\right)_{\text{НВ}} = \frac{0.472}{c_{x_0}} c_{y\,\text{opt}},$$

or, since $c_{y\,\text{opt}} = \sqrt{\frac{c_{x_0}}{D}}$,

$$\left(\frac{c_y}{c_x}\right)_{\text{НВ}} = \frac{0.472}{\sqrt{c_{x_0} D_0}}. \qquad (2.14)$$

Substituting (2.14) in (2.9'), we will obtain

$$L_{max} = 4.17 \frac{1}{g c_{p_0}} \frac{0.472}{\sqrt{c_{x_0} D_0}} V_{\text{крейс}} \ln \frac{1}{1 - \bar{G}_{\text{т.крейс}}}. \qquad (2.15)$$

either

$$L_{max} = 7.0 \sqrt{p_0 \bar{P}_0} \frac{1}{c_{p_0} c_{x_0} \sqrt{D_0}} \ln \frac{1}{1 - \bar{G}_{\text{т.крейс}}},$$

or

$$L_{max} = 584 \frac{1}{\psi c_{p_0}} \frac{M_{\text{крейс}}}{\sqrt{c_{x_0} D_0}} \ln \frac{1}{1 - \bar{G}_{\text{т.крейс}}}, \qquad (2.15')$$

where cruising speed

$$V_{\text{крейс}} = 12.8 \sqrt{\frac{p_0 \bar{P}_{0\xi}}{c_{x_0}}} \text{ km/h} \qquad (2.16)$$

and corresponding to it number

$$M_{\text{крейс}} = 0.012 \sqrt{\frac{p_0 \bar{P}_{0\xi}}{c_{x_0}}}. \qquad (2.16')$$

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Formulas (2.16) and (2.16') can be obtained via substitution in (2.3) and (2.3') values $c_{x\,\text{вв}} = 1.5\ c_{x_0}$.

Using calculated or experimental data for ψ, c_{p0}, c_{x0} and D_0 for given value $\bar{G}_{\text{т.крейс}}$, it is possible with the help of formula (2.15) to obtain dependence L_{max} on $M_{\text{крейс}}$ (Fig. 2.7).

Distance of takeoff/run-up $L_{\text{разб}}$ during the takeoff.

Distance of takeoff/run-up $L_{\text{разб}}$ is expressed by following formula [13]:

$$L_{\text{разб}} = \frac{1}{2g} \int_0^{V^2_{\text{отр}}} \frac{dV^2}{\bar{P} - f - \frac{q}{p_0}(c_x - fc_y)}. \qquad (2.17)$$

If we take thrust-weight ratio $\bar{P}$ on takeoff/run-up constant and different starting thrust-weight ratio $\bar{P}=\bar{P}_0$, and expression $c_x - fc_y = 0$ in view of its smallness, then integral (2.17) can be calculated analytically in the form

$$L_{\text{разб}} = \frac{1}{2g} \frac{V_{\text{отр}}^2}{\bar{P}_0 - f}.$$

Since the value of velocity head during breakaway

$$q_{\text{отр}} = \frac{\rho V_{\text{отр}}^2}{2} = \frac{V_{\text{отр}}^2}{16},$$

with

$$c_y = \frac{p_0}{q} = \frac{16 p_0}{V_{\text{отр}}^2},$$

that

$$V_{\text{отр}}^2 = \frac{16 p_0}{c_{y\,\text{отр}}}$$

and

$$L_{\text{разб}} = \frac{0.82 p_0}{c_{y\,\text{отр}} (\bar{P}_0 - f)}. \qquad (2.17')$$

or with afterburning

$$L_{\text{разб}} \approx \frac{0.82 p_0}{c_{y\,\text{отр}} (\bar{P}_{0\Phi} - f)}.$$

Dependence of flight characteristics on the parameters of aircraft. c_x, c_{x0}, D_0, $\bar{P}_0$, p_0 and ξ

Examining formulas (2.3), (2.3'), (2.4), (2.4'), (2.5), (2.7), (2.16), (2.16') (2.17), (2.17'), we can make conclusion that values of fundamental flight characteristics V_{max}; $V_{\text{крейс}}$; L_{max}; V_y; $H_{\text{пот}}$; $L_{\text{разб}}$ depend on following parameters and coefficients: c_x and c_{x0}; D_0; $\bar{P}_0$, p and ξ.

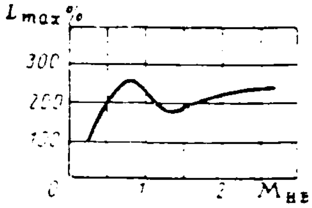

Fig. 2.7. Dependence of relative distance $\bar{L}_{вз}$ on number $M_{вз}$

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Let us consider each of given parameters and will try to establish/install, on what factors their value depends and how they affect fundamental aircraft performance.

Drag coefficient c_x corresponds to total aerodynamic drag of aircraft X and can be expressed as follows:

$$c_x = \frac{X}{qS} = \frac{X_{f} + X_{в.кр} + X_{оп} + X_{в.оп} + X_{ф} + X_{в.ф} + X_{г} + X_{в.г} + X_{i} + X_{н}}{qS},$$

where q - velocity head of incident flow;

X_{f} - shape drag and friction of wing;

$X_{в.кр}$ - wave wing drag;

$X_{оп}$ - shape drag and friction of tail assembly;

$X_{в.оп}$ - matched impedance of tail assembly;

$X_{ф}$ - shape drag and friction of fuselage;

$X_{в.ф}$ - matched impedance of fuselage;

$X_{г}$ - shape drag and friction of engine nacelles, nacelles of chassis/landing gear and so forth;

$X_{в.г}$ - wave pod drag of engines, nacelles of chassis/landing gear;

X_{i} - induced drag, i.e., resistance, which depends on angle of attack or c_y

$X_{\text{в}}$ - interference drag, which appears as a result of mutual wing influence, fuselage, tail assembly, engine nacelles with streamlining.

Using appropriate dimensionless coefficients, let us register coefficient of total drag of aircraft in the following form:

$$c_x = c_{xp} + c_{x\,\text{в.р}} + (c_{x\,\text{оп}} + c_{x\,\text{в.оп}})\frac{S_{\text{оп}}}{S} + (c_{x\,\text{ф}} + c_{x\,\text{в.ф}})\frac{S_{\text{м.ф}}}{S} + (c_{x\,\text{г}} + c_{x\,\text{в.г}})\frac{S_{\text{м.г}}}{S} + c_{xi} + c_{x\,\text{н}},$$

where $c_{x\,\text{оп}}$; $c_{x\,\text{в.оп}}$ - drag coefficients, in reference to area of tail assembly $S_{\text{оп}}$;

$c_{x\,\text{ф}}$; $c_{x\,\text{в.ф}}$; $c_{x\,\text{г}}$; $c_{x\,\text{в.г}}$ - drag coefficients, in reference to appropriate areas of midsection of fuselage $S_{\text{м.ф}}$ and nacelles $S_{\text{м.г}}$.

It is convenient for analysis to represent coefficient of total drag of aircraft c_x in the form of sum of two coefficients c_{x0} and c_{xi}, from which

$$c_{x0} = c_{xp} + c_{x\,\text{в.р}} + (c_{x\,\text{оп}} + c_{x\,\text{в.оп}})\frac{S_{\text{оп}}}{S} + (c_{x\,\text{ф}} + c_{x\,\text{в.ф}})\frac{S_{\text{м.ф}}}{S} + (c_{x\,\text{г}} + c_{x\,\text{в.г}})\frac{S_{\text{м.г}}}{S} + c_{x\,\text{н}}$$

it corresponds to drag of aircraft, which it has when $c_y=0$, and $c_{xi}=D_0c_y^2$ it corresponds to further wing drag, which it is supplemented when $c_y\neq 0$. In this case it is considered that the further resistance, which appears in tail assembly, fuselage, nacelles as a result of changing the angle of attack, is negligibly small.

Thus, drag coefficient of aircraft c_x can be with sufficient approximation/approach examined by that consisting of two parts $c_x = c_{x0} + c_{xi}$.

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From aerodynamics it is known that total frontal aerodynamic drag of aircraft changes according to Mach number both in its total quantity and on relationship/ratio between components of separate forms of resistance: form, friction, wave, interference and inductive.

Graph Fig. 2.8 shows character of change in coefficient c_x in dependence on Mach number in aircraft with various forms (on graph to each value c_x it corresponds its value $c_y=\frac{p}{q}$. Entire speed range of contemporary aircraft in the dependence on the character of change c_x can be decomposed into three zones: the first (from M=0) to $M=M_{крит}$) - subsonic, the second (from $M_{крит}$ to M≈1.2) - transonic, the third (from M=1.2 to M=5) - supersonic.

Let us consider how changes relationship/ratio between components c_x in each of zones pointed out above.

In the first, subsonic, to zone drag, which does not depend on angle of attack of wing, to which corresponds coefficient c_{x0} includes: shape drag, friction, interference drag. To this resistance is supplemented the resistance inductive, which depends on the angle of attack (from c_y), to which corresponds coefficient c_{xi}:

$$c_{xi}=D_0c_y^2=D_0\frac{p^2}{q^2}=\frac{256p^2}{(aM)^4\Delta\pi\lambda}.$$

From this formula it is evident that with assigned magnitudes p, λ and Δ with decrease of M c_{xi} considerably is increased (Fig. 2.9).

From the resistance, to which corresponds c_{x0}. large value in this zone has frictional resistance. Therefore for the low-speed aircraft, which possess maximum speed in the first zone, for decrease c_x should be applied wings with the sufficiently large aspect ratio λ and accepted the moderate specific wing loads. For decreasing the frictional resistance for similar aircraft is expedient the application of such forms of its parts, with which would be ensured the low value of accelerating pressure gradients on the surfaces, streamlined with flow. The latter fact contributes to the retention/maintaining laminar boundary layer in the large sections of the surface of streamlined bodies and, consequently, it leads to the decrease of frictional resistance.

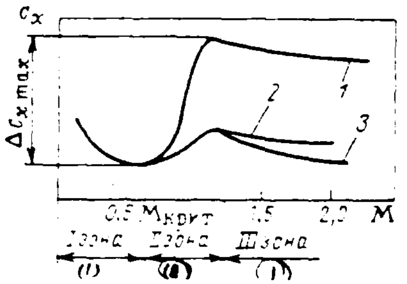

Fig. 2.8.

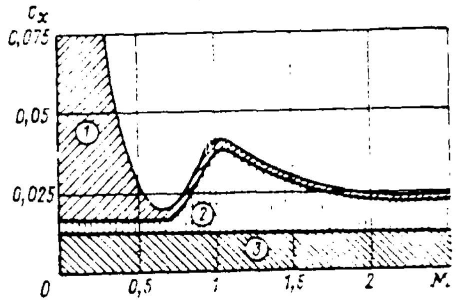

Fig. 2.9.

Fig. 2.8. Change c_x of aircraft in dependence on Mach number: 1 - subsonic aircraft; 2 - supersonic aircraft with rounded wing leading edge; 3 - supersonic aircraft with pointed wing leading edge.

Key: (1). zone.

Fig. 2.9. Character of change in components of coefficient total drag of aircraft in dependence on Mach number for medium altitudes (H≅10000 m, p=250 kgf/m², λ=3.2): 1 - induced drag, which depends on spread/scope; 2 - shape drag and matched impedance; 3- frictional resistance.

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For this is necessary the application of the laminated profiles of wings and laminated enclosures of fuselages and nacelles. However, one should remember that for the laminated profiles/airfoils lowered/reduced values $c_{y\,max}$ are characteristic therefore gain in the frictional resistance can be considerably lowered as a result of the need for increase S for the satisfaction of the requirements, presented to takeoff-landing characteristics.

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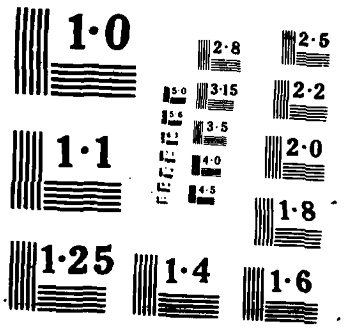
1·0
2·8
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4·5
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1·4
1·6

Secondly, transonic, to zone resistance, to which corresponds coefficient c_{x0}. in reaching/achievement of Mach number, called critical $M_{kpит}$ as a result of emergence and violent increase in matched impedance, sharply increases (see Fig. 2.8). For the subsonic aircraft with the subsonic forms of wing, fuselage, nacelles and tail assembly this increase can be in ten or more times. An increase in the resistance occurs sufficiently sharply; therefore the phenomenon was called characteristic name "sound barrier". For the aircraft, the value of maximum speed of which is located in the second zone, is expedient the application of forms, which facilitate the decrease of an increase in the matched impedance (Fig. 2.10), i.e. sweptback wings and tail assembly, wings and the tail assembly of low elongation/aspect ratio (with M>1), especially triangular planform, and also application for the wings and the tail assembly of profiles with small relative thickness ($\bar{c}$=0.06-0.08) and with the low concavity also of fuselages with the great lengthening.

Application of these forms can ensure comparatively small value of increase $\Delta c_{x\max}$ (see Fig. 2.8).

In the third, the supersonic, to zone the matched impedance, caused by bow shocks, comprises considerable portion in total drag of the aircraft, to which $c_x=c_{x0}+c_{xi}$ corresponds. Imparting to the parts of the airplane wing, to fuselage, to tail assembly and so forth - special supersonic forms, which ensure the emergence of oblique bow waves, i.e., the application of thin wing profiles and tail assembly ($\bar{c}$=0.03-0.05) with the pointed leading edges, the enclosures of fuselages with the pointed, strongly elongated nose section and the great lengthening and the like (Fig. 2.11) significantly reduces the matched impedance of aircraft.

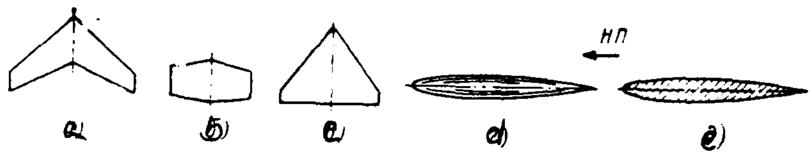

Fig. 2.10. Wing planform, its profile/airfoil and form of fuselage, that are used on transonic aircraft ($M \cong 1.0$): a) sweptback wing; b) tapered low-aspect-ratio wing; c) delta low-aspect-ratio wing; d) fuselage with great lengthening; e) wing profile (laminar), relative thickness $\bar{c}$=0.10.

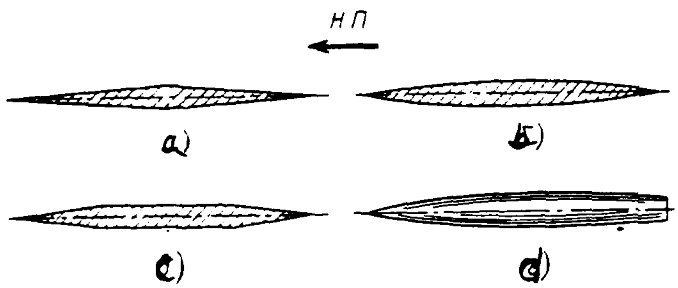

Fig. 2.11. Forms of wing profiles (a, b and c) and fuselage (d), appropriate for supersonic speeds ($M>1.5$).

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However, one should consider that the wing profiles with the sharp leading edges possess unsatisfactory properties with respect to the landing data of aircraft and therefore to use them without the special high-lift device of wing (drooped noses) does not follow. In the supersonic zone is relatively great the frictional resistance, not value of which, as is known, affects the smoothness of the doomed body surface and therefore wings and supersonic fuselage must have least possible surface roughness. For the supersonic aircraft the application of low-aspect-ratio wings (λ<2.5) is expedient, since the latter on the supersonic aircraft plays insignificant role in increase c_{xi} On the other hand, the wings of such type make it possible in the supersonic range to reduce matched impedance. Taking into account during the design effect on value c_x and c_{x0} of the examined above factors, designer must remember also that value c_{x0} depends on the specific wing load p_0. Let us assume that are planned completely similar aircraft of one and the same weight G_0, but with wings $(G_0=\text{const};\ S_{кр}=\text{var})$ different in the area.The areas of maximum cross sections of fuselage and engine nacelles ($S_{м.ф}$ and $S_{м.г}$) will be identical, and relationship of the areas of tail assembly and area of their wings are constant $S_{оп}/S=\text{const}$.

It is possible also to consider that

$$\frac{c_{x\,оп}+c_{x\,в.оп}}{c_{x\,кр}+c_{x\,в.кр}}=\gamma=\text{const}$$

and

$$c_{x\,ф}+c_{x\,в\,ф}=c_{x\,г}+c_{x\,в\,г}.$$

Furthermore, for similar aircraft is correct the equality

$$\frac{S_{м.ф}}{S}+\frac{S_{м.г}}{S}=\frac{p_0}{G_0\,\Sigma S_м}=\frac{p_0}{k_1},$$

where $\Sigma S_м$ - sum of the areas of maximum cross sections of fuselage and nacelles of engines

$$k_1=\frac{G_0}{\Sigma S_м}.$$

Then assuming/setting

$$1+\gamma\frac{S_{on}}{S}=k_0$$

and disregarding coefficient $c_{x\text{в}}$, we can write

$$c_{x0}=(c_{xp}+c_{x\,\text{в.кр}})k_0+(c_{x\,\Phi}+c_{x\,\text{в.}\Phi})\frac{p_0}{k_1}.$$

For subsonic speeds $(c_{x\,\text{в.кр}}=0;\ c_{x\,\text{в.}\Phi}=0)$ will be applicable the following formula:

$$c_{x0}=k_0c_{xp}+c_{x\,\Phi}\frac{p_0}{k_1}{}^{*}, \qquad (2.18)$$

while for supersonic speeds can be used approximate formula

$$c_{x0}\cong 5.0k_0\bar{c}^2\frac{1}{\sqrt{M^2-1}}+c_f+c'_{x\Phi}\frac{p_0}{k_1}, \qquad (2.18')$$

where $c'_{x\Phi}=c_{x\,\Phi}+c_{x\,\text{в.}\Phi}$.

FOOTNOTE [1]. In the practice of preliminary (sketch) design c_{x0} it is determined either by calculation according to the approximation methods, or from the blasting of similar according to airplane design. ENDFOOTNOTE.

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On the basis of formulas (2.18) and (2.18') it is possible to judge effect not c_{x0} of specific wing load p_0. Formula (2.18') gives representation about the considerable effect on value c_{x0} at supersonic speeds of the relative thickness of airfoil of wing $\bar{c}$, which enters into formula to the second degree.

Coefficient D_0, entering formula, sufficiently precise for flight speed range of aircraft

$$c_x = c_{x0} + D_0 c_y^2 = c_{x0} + D_0 \frac{p^2}{q^2}.$$

is second coefficient, which characterizes aerodynamic properties of aircraft Value D_0 depends on different factors depending on that, what zone includes the speed of the projected/designed aircraft. In the first zone (subsonic) coefficient D_0[1] is equal to

$$D_0 = \frac{k_2}{\pi \lambda_{эф}},$$

where k_2=1.02 for the tapered wings with the elongation/aspect ratio $\lambda>3$; k_2=1.6 for the delta wings with the elongation/aspect ratio $\lambda\approx2$.

FOOTNOTE [1]. Let us recall that $D_0 = 1/c_y^{a}$. ENDFOOTNOTE.

Consequently, as it was already noted above, for aircraft, whose speeds can be related to first zone, must be used average/mean (λ=4-7) and large (λ=7-12) wing aspect ratios. Is especially important the application of large λ for the aircraft with the long range of flight.

In supersonic, the third, to zone coefficient D_0 can be expressed thus:

$$D_0 = 0.25 B_0 \sqrt{M^2 - 1}.$$

Coefficient B_0 is expressed by the following formulas:

for the straight/direct tapered wing

$$B_0 = \frac{1}{1 - \frac{1}{2\lambda\sqrt{M^2 - 1}}};$$

for the delta wing with the supersonic leading edges

$$B_0 = 1;$$

for the delta wing with the subsonic edges

$$B_0 = \frac{1}{\pi}\left(\frac{4}{\lambda\sqrt{M^2-1}} + \frac{\lambda\sqrt{M^2-1}}{1,8}\right).$$

Thus, D_0 at subsonic and transonic speeds (M<1) is expressed dependence of aerodynamic drag when $c_y=0$ on wing planform (λ and η), and at supersonic speeds - and on wing planform and speed (M). Consequently, designer can attain decrease c_{x0} and D_0 by the adjustment of the aerodynamic shape of wing, fuselage, tail assembly of aircraft. In this case, as we saw above, for each speed range were characteristic special shapes of profiles/airfoils, wing in the plan/layout, fuselage, etc. However, decrease c_{x0} and D_0 (with the constant values of other parameters), as it follows from the examination of formulas (2.3); (2.3'); (2.5); (2.6); (2.15); (2.15'); (2.16); (2.16'), leads to an improvement in the aircraft performance

$$V_{max}(M_{max});\ V_{крейс}(M_{крейс});\ H_{пот},\ V_{y\,max},\ L_{max}.$$

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Following parameter, considerably affecting all fundamental flight characteristics, is thrust-weight ratio

$$\bar{P} = \frac{P}{G_0}$$

or in question during design starting (for TRD) or initial (PVRD) thrust-weight ratio

$$\bar{P}_0 = \frac{P_0}{G_0}.$$

It is easy to see that expression (2.19) has definite meaning only in such a case, when are determined P_0 and G_0. However, with piloting of aircraft these values must be found, moreover task is complicated by the fact that G_0 proves to be the value, dependent on P_0, and P_0 in turn, depends on G_0. Thus, for the projected/designed aircraft to determine $\bar{P}_0$ according to formula (2.19) is impossible. If we express $\bar{P}_0$, after using formulas (2.3') and (2.16'), then

$$\bar{P}_{0\text{п}} = \frac{4650 M^2_{max} c_x}{\xi p_0}; \qquad (2.20)$$

$$\bar{P}_{0\text{п}} = \frac{6950 M^2_{\text{крейс}} c_{x_c}}{\xi p_0} \qquad (2.20')$$

they will determine those values of thrust-weight ratio, which need for obtaining respectively given ones M_{max} and $M_{\text{крейс}}$ with given ones [illegible] p_0 and ξ. The thrust-weight ratio, expressed (2.20) and (2.20'), we will call required thrust-weight ratio. After determining the value of required thrust-weight ratio $\bar{P}_{0\text{п}}$ which ensures assigned magnitude M_{max} or $M_{\text{крейс}}$ designer must know well the ways, which lead to equality $\bar{P}_{0\text{п}}$ and thrust-weight ratio of "that arranged/located" $\bar{P}_{0\text{р}}$, i.e. thrust-to-weight ratio, which it is possible to virtually ensure with the values of the parameters of aircraft and power plant accepted and at the prescribed/assigned structural strength. The guarantee of equality of value $\bar{P}_{0\text{р}}$ to value $\bar{P}_{0\text{п}}$ is one of the basic tasks of designing the aircraft, which the designer as the final result must solve in the process of design. In order to know, how it is possible to influence value $\bar{P}_{0\text{р}}$, it is necessary to explain the dependence of available thrust-weight ratio $\bar{P}_{0T}$ on different parameters, for which we will use the equation of the over-all payload ratios of the aircraft

$$\overline{G}_{к}+\overline{G}_{с.у}+\overline{G}_{т}+\overline{G}_{с.э.т}=1.$$

We will obtain for second and third members of left side of equation $\overline{G}_{с.у}$ and $\overline{G}_{т}$ equations, that establish/install their dependence on different parameters.

For $\overline{G}_{с.у}$ it follows from determination

$$\overline{G}_{с.у}=\frac{G_{с.у}}{G_0}=\frac{G_{с.у}P_0}{G_0P_0}=r_0\overline{P}_0. \qquad (2.21)$$

where $r_0=\frac{G_{с.у}}{P_0}$ - specific weight of power plant of aircraft (referred to boost for launching P_0).

For over-all payload ratio of fuel/propellant $\overline{G}_{т}$ it is possible to obtain dependence on parameters as follows. The complete fuel reserve on the aircraft, to which corresponds over-all payload ratio $\overline{G}_{т}$, consists: from the fuel/propellant, expended in cruise $G_{т.крейс}$, the fuel/propellant, required to the taxiing, takeoff run, the takeoff, lift and landing $G_{т.п}$, the fuel/propellant, required on dispersal/acceleration $G_{т.р}$ and navigational reserve $G_{т.н.з}$.

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Consequently, the complete reserve

$$G_{т}=G_{т.крейс}+G_{т.п}+G_{т.р}+G_{т.н.з}.$$

Passing to the over-all payload ratios, we will have

$$\overline{G}_{т}=\overline{G}_{т.крейс}+\overline{G}_{т.п}+\overline{G}_{т.р}+\overline{G}_{т.н.з}.$$

For aircraft with TRD and PVRD the overall payload ratio of the fuel/propellant, expended to the cruise (with given one L_{max}), we will obtain, solving equation (2.15) relatively $\bar{G}_{т.крейс}$: $\bar{G}_{т.крейс} = 1 - e^{-\frac{L_{max}}{A}}$,

where
$$A = 7.0 \frac{1}{c_{f0}} \sqrt{p_0 \bar{P}_0 \xi} \frac{1}{c_{x_0} \sqrt{D}} = \frac{5840 M_{крейс}}{c_{f0} \sqrt{c_{x_0} D_0}}.$$

or for the linearized formula (replacement of logarithmic curve by two straight lines)

$$\bar{G}_{т.крейс} = R c_{f0} \frac{L_{max} \sqrt{c_x D}}{M_{крейс}} - u.$$

where u - coefficient, depending on values $\bar{G}_{т.крейс}$; $R = 0.00145$ and u=0 when $\bar{G}_{т.крейс} \leqslant 0.3$, $R = 0.00100$ and u=0.09 when $0.5 > \bar{G}_{т.крейс} > 0.3$. With the prescribed/assigned duration of flight t ' will be real the formula (for H, which corresponds to Δ) $\bar{G}_{т.крейс} = \xi \bar{P}_0 \Delta t'$. Knowing $\bar{G}_{т.крейс}$ it is possible to determine further $\bar{G}_{т.н}$, $\bar{G}_{т.р}$ and $\bar{G}_{т.к}$: $\bar{G}_{т.н} = 0.0009 H_{крейс}$, where $H_{крейс}$ they determine after finding

$$\Delta_{крейс} = \frac{1.76 \sqrt{c_x D}}{\xi \bar{P}}.$$

With large numbers M ≥2 over-all payload ratio of fuel reserve, required to dispersal/acceleration $G_{т.р}$ is determined from formula

$$\bar{G}_{т.р} = c_{f0} \xi \bar{P}_0 \Delta t', \text{ where } t' = \frac{1}{g} \int_{V_1}^{V_2} \frac{dV}{\bar{P}_{расп} - \bar{P}_{потр}},$$

and Δ corresponds to altitude of dispersal/acceleration. For the aircraft of subsonic and transonic $\bar{G}_{т.р}$ very little and can be disregarded.

Over-all payload ratio of navigational fuel reserve is determined from approximation formula [30]

$\bar{G}_{т.з}=0.10\bar{G}_{т.крейс}$ (for military aircraft);

$\bar{G}_{т.з}=(0.15-0.20)\bar{G}_{т.крейс}$ (for passenger aircraft).

Thus, for over-all payload ratio of fuel/propellant on aircraft $\bar{G}_т$ can be registered following formulas.

If flying range is prescribed/assigned, then

$$\bar{G}_т=S\left(1-e^{-\frac{L_{max}}{A}}\right)+u, \qquad (2.22)$$

where $S=1,1;\ u=0,0009H_{крейс}+\xi c_{p0}\bar{P}_0\Delta t'$ (if $M_{крейс}\geqslant 2,0$);

$S=1,15-1,2;\ u=0,0009H_{крейс}$ (if $M_{крейс}<2,0$).

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If $\bar{G}_{т.крейс}\leqslant 0,3$, then

$$\bar{G}_т=SR\xi c_{p0}\frac{L_{max}\sqrt{c_{x_0}D_K}}{M_{крейс}}+u, \qquad (2.23)$$

where $S=1.1;\ u=0.0009H_{крейс}+\xi c_{p0}\bar{P}_0\Delta t'$ (if $\bar{G}_{т.крейс}\leqslant 0,3$ and $M_{крейс}\geqslant 2,0$);
$S=1.15-1.2;\ u=0,0009H_{крейс}$ (if $\bar{G}_{т.крейс}\leqslant 0,3$ and $M_{крейс}<2.0$);
$u=0,0009H_{крейс}+\xi c_{p0}\bar{P}_0\Delta t'-0,09$ (if $\bar{G}_{т.крейс}>0,3$ and $M_{крейс}>2.0$); $u=0.0009H_{крейс}+0.09$ (if $\bar{G}_{т.крейс}>0.3$ and $M_{крейс}<2.0$).

If duration of flight t' is prescribed/assigned, then

$$\bar{G}_т=S\xi\bar{P}_0\Delta c_{p0}t'+u, \qquad (2.23')$$

where $S=1.1;\ u=0.0009\ H_{крейс}$.

For aircraft with TVD it is possible to use formula

$$\overline{G}_{т} = 0{,}000057 S L_{\max} c_e \sqrt{\frac{c_{x_c}}{t}}, \qquad (2.23'')$$

where S = 1.15-1.2.

Using formulas (2.21), (2.23) and (2.23') and equation of overall payload ratios, we will obtain following formulas for available thrust-weight ratio $\bar{P}_{0р}$.

if is prescribed/assigned duration of flight t′, then

$$\overline{P}_{0р} = \frac{1 - \overline{G}_{к} - \overline{G}_{c.у}}{r_0 - S \cdot 3 c_p \Delta t'}, \qquad (2.24)$$

and if is prescribed/assigned flying range $L_{\max}$, then

$$\overline{P}_{0_р} = \frac{1 - \overline{G}_{к} - \overline{G}_{c.у}}{r_0} - \frac{1}{r_0}\left[\frac{S F c_p L_{\max} \sqrt{c_{x_c} D}}{M_{кр}} + u\right]. \qquad (2.24')$$

These formulas give clear representation about the dependence of available thrust-weight ratio $\bar{P}_{0р}$ on the series/row of design parameters.

For determining quantitative effect of change in one or the other parameter to available thrust-weight ratio $\bar{P}_{0р}$ can be used following formulas:

$$\overline{P}_0' = \overline{P}_0 \frac{1 - \varepsilon_1 \overline{G}_{к}}{1 - \overline{G}_{к}}; \quad \overline{P}_0 = \overline{P}_0 \frac{1 - \varepsilon_2 \overline{G}_{т}}{1 - \overline{G}_{т}};$$

$$\overline{P}_0 = \overline{P}_0 \frac{1 - \varepsilon_3 \overline{G}_{т}}{1 - \overline{G}_{т}}; \quad \overline{P}_0 = \frac{\overline{P}_0}{1 - \overline{G}_{c.у}(\varepsilon_4 - 1)},$$

where $$\varepsilon_1 = \frac{\overline{G}_{к}'}{\overline{G}_{к}}; \quad \varepsilon_2 = \frac{c_{p0}'}{c_{p0}}; \quad \varepsilon_3 = \frac{L_{\max}'}{L_{\max}}; \quad \varepsilon_4 = \frac{r_0'}{r_0};$$

$\overline{P}_0'$ - new value $\overline{P}_0$, obtained, if we $\overline{G}_{к}$ increase on $\Delta\overline{G}_{к}$.

If we increase thrust-weight ratio $\overline{P}_0$ by installation on aircraft of two engines instead of one, then when $L_{max}=\text{const}$

$$G_{c.э.г}=\text{const} \text{ and } \overline{G}_{к}'=\varepsilon_1\overline{G}_{к};$$
$$\overline{P}_0=\frac{1-(\overline{G}_{к}\varepsilon_1-\overline{G}_{т})}{\frac{0,5\overline{G}_{c.э.г}}{\overline{P}_0}+r_0\varepsilon_4}.$$

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Formulas given above easily can be obtained from equation of over-all payload ratios, which we will write in removed/abstracted form $\overline{G}_1+\overline{G}_2+\overline{G}_3+\overline{G}_4=1$; if we increase $\overline{G}_1$ on $\Delta\overline{G}_1$, then $\overline{G}_2$; $\overline{G}_3$ and $\overline{G}_4$ will be changed and equation will be registered as follows:

$$\overline{G}_1+\Delta\overline{G}_1+\overline{G}_2'+\overline{G}_3'+\overline{G}_4'=1.$$

Then

$$\overline{G}_i'=\overline{G}_i\left(1-\frac{\Delta\overline{G}_1}{1-\overline{G}_1}\right);\quad \Delta\overline{G}_i=-\overline{G}_i\frac{\Delta\overline{G}_1}{1-\overline{G}_1},$$

but

$$G_0'=G_0\frac{1}{1-\frac{\Delta\overline{G}_1}{1-\overline{G}_1}}.$$

Assuming that P'=P, we will obtain

$$\overline{P}_0'=\overline{P}_0\left(1-\frac{\Delta\overline{G}_1}{1-\overline{G}_1}\right),$$

or generally

$$\overline{P}_0'=\overline{P}_0\left(1-\frac{\Delta\overline{G}_i}{1-\overline{G}_i}\right).$$

Examining formulas (2.24) and (2.24'), it is possible to do conclusion that equality to thrust- weight ratio of that arranged/located $\bar{P}_{0p}$ and thrust-weight ratio of required $\bar{P}_{0n}$ with given values of all other parameters, entering formulas indicated, possibly only with one certain required value of over-all payload ratio of construction/design of aircraft $\bar{G}_{\kappa.n}$, expression for which we will obtain from formulas (2.24) and (2.24'):

$$\bar{G}_{\kappa.n}=1-\bar{G}_{c.э.r}-\bar{G}_{0p}(r_0+S\xi c_{p0}t').$$

or

$$\bar{G}_{\kappa.n}=1-\bar{G}_{c.э.r}-\bar{P}_{0p}r_0-SR\xi c_{p0}\frac{L_{max}\sqrt{c_{x_0}D_0}}{M_{\kappa p}}-u.$$

If aircraft with given parameters and characteristics to carry out possibly, then equality

$$\bar{P}_{0p}=\bar{P}_{0n}.$$

will be satisfied According to formulas (2.20) and (2.20') we can find the value of the over-all payload ratio of construction/design $\bar{G}_{\kappa.n}$. required for guaranteeing the assigned magnitudes M_{max}, $M_{\kappa p}$ and L_{max} in prescribed/assigned parameters c_{x0}, p_0, c_{p0} and other,

$$\bar{G}_{\kappa.n}=1-\bar{G}_{c.э.r}-\frac{4650M_{max}^2c_x}{\xi p_0}(r_0+S\xi\Delta c_{p0}t'); \qquad (2.25)$$

with preset time t' of flight or

$$\bar{G}_{\kappa.n}=1-\bar{G}_{c.э.r}-\frac{6950M_{\kappa p}^2c_{x_0}}{\xi p_0}r_0-SR\xi c_{p0}\frac{L_{max}\sqrt{c_{x_0}D_0}}{M_{\kappa p}}-u \qquad (2.25')$$

or by given one L_{max}

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Available over-all payload ratio of construction/design $\bar{G}_{\kappa.r}$ can be expressed approximately by formula (2.26) for aircraft with straight/direct or sweptback wing of large or average/mean elongation/aspect ratio

$$\overline{G}_{\kappa p} = (0.027\varphi\mu n_A \frac{G_0^{1/2}}{\cos\chi} \sqrt{\frac{\lambda}{p_0} + \frac{5.5}{p_0}})(1 + \beta_1\lambda_\phi m + \beta_2) - 0.065 \quad (2.26)$$

or for delta-wings airplane of low elongation/aspect ratio

$$\overline{G}_{\kappa p} = \left(0.049\varphi\mu n_A G_0^{1/2} \sqrt{\frac{\lambda}{p_0} + \frac{5.5}{p_0}}\right)(1 + \beta_1\lambda_\phi m + \beta_2) + 0.065. \quad (2.26')$$

Here $\varphi = 1 - \frac{3(\eta - 1)}{\eta - 2}(\overline{z}_1\varepsilon_1\overline{G}_T + \overline{z}_2\varepsilon_2\overline{G}_{c.y})$ -coefficient of unloading wing,

where η - wing taper;

ϵ_1 - portion of fuel/propellant, arranged/located in wing;

$\overline{z}_1$ - relative, in portions of semispan, coordinate of center of gravity of fuel in wing (relative to axis of symmetry of aircraft);

ϵ_2 - portion of weight of power plant in wing;

$\overline{z}_2$ - relative coordinate in portions of semispan center of gravity. power plant;

$\mu = 1 - \varepsilon\left(\frac{\sigma_T}{\sigma'_T} - 1\right)$ - coefficient of the weight increase of the construction/design of aircraft as a result of the account of kinetic heating;

ϵ - ratio of the weight of the load-bearing loaded in flight elements to the weight of the entire structure of the aircraft (in the first approximation, it is possible to accept ϵ=0.5);

$\frac{\sigma_T}{\sigma'_T}$ - relation of yield points at a normal temperature and during the kinetic heating;

n_A - coefficient of calculated g-force;

G_0 - gross weight of aircraft in t;

β_1=0.07- 0.09 - for the supersonic aircraft;

β_1=0.065-0.08 - for the heavy subsonic and transonic aircraft;

β_1=0.08-0.115 - for the transonic transport aircraft;

m=1 - for the supersonic aircraft;

m=1.2-1.3 - for the subsonic and transonic aircraft;

β_2=0.27 - for the supersonic aircraft;

β_2=0.15 - for the subsonic and transonic aircraft;

λ_{ϕ} - fineness ratio of fuselage.

For aircraft of specific type and with known unloading of wing can be used approximation simplified formula (2.26")

$$\overline{G}_{\kappa p}=\beta\frac{G_0^{1.2}}{p_0^{1.2}}+\frac{15}{p_0}+0{,}065, \qquad (2.26'')$$

where G_0 - in t;

β=1.6 - for fighters;

β=0.7-0.8 - for passenger aircraft with two TVD with unloading of wing;

β=0.4-0.5 - for passenger aircraft with four TVD with large unloading of wing [1];

β=0.55 - for passenger aircraft with two TRD with unloading of wing only by fuel/propellant;

β=0.35 - for passenger aircraft with four TRD with large unloading of wing;

β=0.35 - for carriers with four TRD with large unloading of wing.

FOOTNOTE [1]. Large unloading - unloading wing with a large quantity of loads (engines, fuel/propellant). The low value of the coefficient of unloading φ corresponds to large unloading. ENDFOOTNOTE.

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Obviously, designer in process of design, choosing these or other values of parameters, entering formulas (2.26), (2.26'), can reach equality of available over- all payload ratio of construction/design $\bar{G}_{\kappa}$ required value $\bar{G}_{\kappa.n.}$ thereby having successfully solved basic task of design.

Let us consider how affects increase in thrust- weight ratio $\bar{P}_0$ on basic flight characteristics V_{max}; M_{max}; $V_{\kappa p}(M_{\kappa p})$; H_{n}; L_{max}; $V_{\mu\, max}$ and L_{p}.

We will use formula (2.20) for graphing of dependence M_{max} on $\bar{P}_0$. Since with increase in $\bar{P}_0$ coefficients c_x and ξ as a result of change M will change, then, being assigned by graphic or computed values M_{max} for each value M_{max} we will obtain c_x and ξ. Substituting these values in (2.20), let us find the appropriate values of $\bar{P}_0$ and as a result will construct the graph/diagram of dependence M_{max} on $\bar{P}_0$ according to formula (2.12). We see that with increase in $\bar{P}_0$ value M_{max} steadily grows. The same can be obtained. also, for $M_{\kappa pe\ddot{u}c}$. Using formula (2.5), the dependence of relative density of air on height/altitude H and taking into account that the flight on the ceiling occurs on maximum quality K_{max} for which

$$c_x = 2c_{x_0}.$$

we can construct the graph/diagram of dependence $H_{пот}$ about $\overline{P}_0$. For this, after writing formula (2.20) for the case of flight at a velocity, which corresponds to Mach number on the ceiling,

$$\overline{P}_0=\frac{4650M^2 \cdot 2c_{x_0}}{\xi p_0}=\frac{9300M^2c_{x_0}}{\xi p_0},$$

being assigned by the series/row of values M, determining for each value by calculation or according to graphs

$$c_{x_0}=f(M);\quad \xi=f(M) \quad \text{and} \quad D_0=f(M)$$

and, after substituting these values into formula (2.5), we will obtain data for the graph

$$H_{п.т}=f(\overline{P}_0),$$

from which evident that with an increase $\overline{P}_0$ ceiling $H_{пт}$ of aircraft increases (Fig. 2.13).

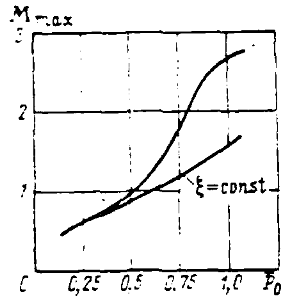

Fig. 2.12.

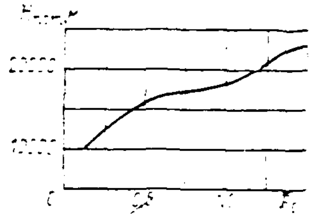

Fig. 2.13.

Fig. 2.12. change in the maximum speed of flight, which corresponds to number M_{max} of aircraft with TRD depending on the starting thrust-weight ratio $\bar{P}_0$ (H=11000 m).

Fig. 2.13. Dependence of ceiling $H_{пот}$ on starting thrust-weight ratio $\bar{P}_0$.

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Value L_{max} thrust-weight ratio $\bar{P}_0$ affects through $M_{кpeйc}$. Substituting into formula (2.15') values $M_{кpeйc}=f(\bar{P}_0)$ and corresponding values c_{x0}, D_0 and ψ_0, it is possible to obtain graph $L_{max}=f(\bar{P}_0)$ (see Fig. 2.7), from which it is evident that distance L_{max} first increases, and then upon the appearance of matched impedance (at transonic speeds) it begins sharply to be reduced, whereas with further increase of $\bar{P}_0$ again it begins to grow to some limit.

In a similar manner it is easy to arrive at conclusion that $V_{y\,max}$ with increase in $\bar{P}_0$ grows, and takeoff run length $L_{paзб}$ with increase in $\bar{P}_0$ is reduced. Designer usually chooses parameter value in the beginning of design on the basis of general/common considerations or special calculations. This size and weight the parameter has a complicated effect on fundamental flight characteristics both directly, entering into the formula, which determines the value of characteristic, and through other parameters ($\bar{P}_0$ and c_{x0}).

Let us consider, first, what effect proves to be change p_0 to given above aircraft performance without taking into account effect of this change to its other parameters ($\bar{P}_0$, c_{x0}) directly.

On the basis of dependences, expressed by formulas (2.18) and (2.19), it is possible to write

$$c_x = k_0 c_{xp} - c_{x\phi}\frac{p_0}{k} - D_0\left(\frac{p_0}{q}\right)^2. \qquad (2.27)$$

After dividing both parts (2.27) on $c_y = \frac{p_0}{q}$, we will obtain expression for the required thrust-weight ratio

$$\bar{P}_0 = k_0 c_{xp}\frac{1}{p_0} + c_{x\phi}\frac{q}{k} + D_0\frac{p_0}{q}, \qquad (2.28)$$

where

$$k_0 = 1 + \gamma\frac{S_{on}}{S}; \quad \gamma = \frac{c_{x\,on} + c_{x\,n.on}}{c_{xp} - c_x},$$

whence

$$p_0^2 + \frac{q}{D_0}\left(c_{x\phi}\frac{q}{k} - \bar{P}_0\right)p_0 + \frac{q^2}{D_0}c_{xp}k_0 = 0,$$

or

$$p_0 = \frac{q}{2D_0}\left(\bar{P}_0 - c_{x\phi}\frac{q}{k}\right) - \sqrt{\frac{q^2}{4D_0}\left(c_{x\phi}\frac{q}{k} - \bar{P}_0\right)^2 - \frac{q^2}{D_0}c_{xp}k_0}. \qquad (2.29)$$

Using formula (2.29), it is possible to construct graph/diagram of dependence q or M on p_0 under condition of independence $\bar{P}_0$ from p_0 (i.e. when $\bar{P}_0$=const with change of p_0) (see Figs. 2.14, 2.15). Curves give the possibility to make following conclusions:

1 increase in p_0 leads to increase/growth M (or q) [1], but to certain limit of p_0, which increases with an increase $\bar{P}_0$;

2) with increase in p_0 intensity of increase/growth M (i.e. $\frac{dM}{dp_0}$) or q is reduced; with an increase $\bar{P}_0$ $\frac{dM}{dp_0}$ it grows;

3) increase p_0 to high values, close ones p_0 (with prescribed/assigned thrust-weight ratio $\bar{P}_0$), irrational, since in this case increase M is small, and excessively high values p_0 can lead to inadmissible increase in landing speed $V_{\text{пос}}$ and distances of takeoff/run-up and landing run.

FOOTNOTE [1]. Besides the specially stipulated cases, everywhere is intended increase p_0 with decrease of S and c_0=const. ENDFOOTNOTE.

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Effect of p_0 on thrust-weight ratio $\bar{P}_0$ is manifested as follows. With increase in p_0 available over-all payload ratio $\bar{G}_i$ [see (2.26), (2.26'), (2.26" (] is reduced, available thrust-weight ratio $\bar{P}_0$ [see (2.24) and (2.24')] increase, and p_0 is shifted/sheared by the high values and the intensity of increase/growth M (or q) it increases. Over-all payload ratio $\bar{G}_{\text{с.з.г}}$ with change $\bar{G}_{\text{к}}$ on - $\Delta\bar{G}_{\text{к}}$ will be increased on $\Delta\bar{G}_{\text{с.з.г}} = -\bar{G}_{\text{с.з.г}} \frac{\Delta\bar{G}_{\text{к}}}{1-\bar{G}_{\text{к}}}$.

To ceiling $H_{\text{пот}}$ or on relative density $\Delta_{\text{пот}}$, corresponding to ceiling $H_{\text{пот}}$, specific load p_0 affects through c_y [see (2.27)] and through $\bar{P}_{0T}$ [see (2.24) and (2.24')] as a result of effect of p_0 on $\bar{G}_{K}$, entering expressions (2.26) and (2.26'). Having this in mind and analyzing formula (2.5)

$$\Delta_{\text{пот}}=\frac{1{,}66\sqrt{c_{x_0}D_0}}{\xi\bar{P}_0},$$

it is possible to do the following conclusions:

1) increase of p_0 leads to increase c_{x0}, since

$$c_{x_0}=k_0c_{xp}+c_{x\ddagger}\frac{p_0}{k_1}.$$

Consequently, in subsonic and transonic aircraft ceiling $H_{\text{пот}}$ with increase in p_0 will be reduced. Since with increase in p_0 the over-all payload ratio of construction/design $\bar{G}_K$ is reduced and respectively increases available thrust-weight ratio $\bar{P}_{0p}$, then the intensity of the decrease of ceiling with increase in p_0 will be small;

2) increase p_0 for the supersonic aircraft leads in certain cases to an increase in static ceiling $H_{\text{пот}}$. This for the following reason occurs. Speed on the ceiling, i.e. when $\left(\frac{c_y}{c_x}\right)_{\max}$, is equal to

$$V_{\text{пот}}=11{,}1\sqrt{\frac{p_0\bar{P}\xi}{c_{x_0}}}\quad\text{or}\quad M_{\text{пот}}=0{,}0104\sqrt{\frac{p_0\bar{P}_0\xi}{c_{x_0}}}.$$

Consequently, increase p_0 with the constant/invariable thrust-weight ratio $\bar{P}_0$ and at the speeds, which correspond TO M>1.5 will lead to an increase in the velocity on ceiling $V_{\text{пот}}$ or $M_{\text{пот}}$ (effect on $V_{\text{пот}}$ increases c_{x0} as show calculations, less than the effect of increase p_0).

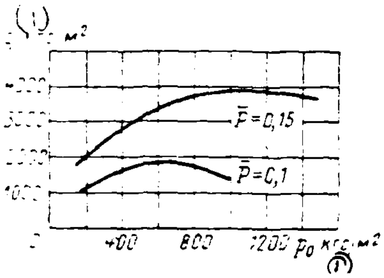

Fig. 2.14.

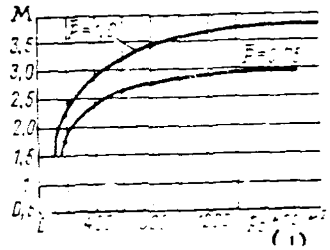

Fig. 2.15.

Fig. 2.14. Effect of velocity head on values p with different values of thrust-weight ratio $\bar{P}$ under conditions, when matched impedance is absent $(\lambda=4,\ c_{x_f}=0.007)$

Key: (1). kgf/m².

Fig. 2.15. Effect of p_0 on value M with different values of thrust-weight ratio $\bar{P}$ of aircraft (H=10000 m).

Key: (1). kgf/m².

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But with an increase in the velocity (with M>1.5) the coefficient ξ of the dependence of the engine thrust will increase to a certain limit in the velocity. This increase with M=1.5-2.5 (but during the boosting TRD and on the large M) will be so/such perceptible, that

$$\Delta_{\text{пот}}=\frac{1{,}161\sqrt{c_x D_\cdot}}{\xi\bar{P}_0}$$

with increase in p_0 and, consequently, increase/growth c_{x0} will be nevertheless reduced, ceiling $H_{\text{пот}}$ will in this case increase. With increase in p_0 the available over-all payload ratio of construction/design $\bar{G}_1$ will be, as we already saw, reduced, available $\bar{P}_0$ will increase, and Δ will be reduced, i.e., H will be increased.

With increase p_0 increases $V_{y\,max}$ - maximum vertical velocity in earth/ground. This is easy to see from the formula

$$V_{y\,max}=1.53\sqrt{\frac{p_0(\bar{P}_0)^3}{c_{x_0}}}.$$

Since $c_x = k_0 c_{xp} + c_{x\phi}\frac{r}{k_0}$, then $\frac{r}{c_x}$ it is possible to register thus: $\frac{1}{\frac{k}{p_0}c_{xp}+\frac{c_{x\phi}}{k_0}}$ from which it is clear that with increase in p_0 fraction $p_0 c_x$ - increases, i.e., $V_{y\,max}$ - increases. $V_{y\,max}$ increases and because is reduced in this case $\bar{G}$ and, consequently, grows available thrust-weight ratio $\bar{P}_{0p}$.

Increase p_0 contributes to increase/growth L_{max}. In this case increase L_{max} will occur to the certain sufficiently high value of p_0, after which the distance will begin to be reduced. For the subsonic and transonic aircraft in the formula

$$L_{max}=7.0\sqrt{p_0\bar{P}_0}\frac{1}{\xi c_p c_x \sqrt{D_0}}\ln\frac{1}{1-\bar{G}_T}$$

of value ξ, ψ and D_0 it is possible to take independent of the velocity (they change they at transonic speed little), then, obviously, L_{max} achieves the greatest value with similar p_0, with which $\sqrt{\bar{P}_0}$ will achieve its maximum value, and $\frac{c_x}{\sqrt{p_0}}$ - minimum value. After making first-order derivative $\frac{c_x}{\sqrt{p_0}}$ on p_0 equal to zero and using formula (2.18), we will obtain

$$p_{0\,\text{вв}}=k_0\frac{c_{xp}}{c_{x\phi}}\frac{G_0}{\Sigma S_{\text{м}}}. \quad (2.30)$$

However, this solution does not consider effect of p_0 on $\bar{G}_K$ and, as a result, to available thrust-weight ratio $\bar{P}_{0p}$. Using graphic methods, it is possible to find p_{OBB} taking into account effect p_0 and $\overline{P}_0$.

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For supersonic aircraft values p_{OBB} in flight to maximum range let us find graphically, after constructing curve of function $L_{\max}$

$$L_{\max}=\frac{584 M_{\text{крейс}}}{\psi c_{p_c} \sqrt{c_{x_c} D_c}} \ln \frac{1}{1-\overline{G}_{\text{т}}}$$

on argument p_0. In this case one should consider that entering the formula values $M_{\text{крейс}}$, c_{x_c}, $\overline{G}_{\text{т}}$ are functions p_0 and ψ, c_p, D_c change in the dependence on M.

After assigning p_0, $M_{\text{крейс}}$ and c_p let us determine, using formulas (2.26") and (2.20') arranged/located $\overline{G}_{\text{кр}}$

$$\overline{G}_{\text{кр}}=3\frac{G_c^{1/2}}{r_c^{1/2}}+\frac{15}{r_c}+0.065,$$

the required thrust- weight ratio $\overline{P}_{\text{пп}}=\frac{69 M^2_{\text{крейс}} c_{x_c}}{p_0}$,

and

$$D_c=0.25 B_c \sqrt{M^2-1}$$
$$\overline{G}_{\text{с.у}}=\overline{P}_c \gamma_c$$

During the satisfactory solution of the problem of design available thrust-weight ratio $\overline{P}_{\text{р}}$ will be equal to required thrust-weight ratio $\overline{P}$ the available over-all payload ratio of fuel/propellant will be

$$\overline{G}_{\text{т}}=1-\overline{G}_{\text{к}}-\overline{G}_{\text{с.у}}-\overline{G}_{\text{снар}}$$

where by value $\overline{G}_{\text{снар}}$ we are assigned.

Then $L_{\max}=\frac{584 M_{\text{крейс}}}{\psi c_p \sqrt{c_{x_c} D_c}}\ln\frac{1}{1-\overline{G}_{\text{т}}}$ and p_0 will serve as the coordinates of the first point of curve $L_{\max}$ and p_0. We are further assigned by the new value of p_0' and, using the taken and calculated above values for $\bar{P}$, $\bar{G}$, $\bar{G}$, let us find

$$\bar{G}_\tau = \bar{G}_k - \Delta\bar{G}_k = \xi\frac{(G_0)^{1.2}}{(F_0)^{1.2}} - \frac{15}{F_0} + 0.065;$$

$$M_{\text{крейс}} = 0.012\sqrt{\frac{F_0\bar{P}_0\xi}{c_{x_0}}};$$

$$c_{x_0} = k_0 c_{xp} + c_{x \text{ в}}\frac{F_0}{k_1}, \quad \bar{P}_0 = \bar{P}_0\frac{1-\varepsilon\bar{G}_k}{1-\bar{G}_k}; \quad \varepsilon = \frac{\bar{G}_k}{\bar{G}_k},$$

where $G = G_0\dfrac{1}{1-\dfrac{\Delta\bar{G}_k}{1-\bar{G}_k}}$.

M we find graphically, after constructing curve $M_{\text{крейс. ист}} = f(M_{\text{крейс. зад}})$ and ray/beam from the origin of coordinates at an angle of 45° to the axes/axles. In this case $M_{\text{крейс. зад}}$ - Mach number, by which we are assigned, choosing values ξ and c_x [1], and $M_{\text{крейс. ист}}$ - Mach number, which we obtain according to the formula for $M'_{\text{крейс}} = 0.012\sqrt{\frac{p_0'\bar{P}_0\xi'}{c_{x_0}}}$. By knowing $M_{\text{крейс}}$ and determining D_0 and $\bar{G}_\tau$ on $\bar{G}_\tau = \bar{G}_\tau\left(1 - \dfrac{\bar{G}_k - \bar{G}_k}{1-\bar{G}_k}\right)$, let us find for p_0'

$$L'_{\max} = \frac{584 M'_{\text{крейс}}}{c_{\xi 0}\sqrt{c_{x_0} D_0}}\ln\frac{1}{1-\bar{G}_\tau}.$$

FOOTNOTE [1]. On curves $\xi = f(M)$ and $c'_{x0} = f(M)$, which are prescribed/assigned or calculated. ENDFOOTNOTE.

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Fig. 2.16 gives curve, constructed thus. In this case an increase in $\bar{G}_{k}$ at large Mach numbers, which is obtained as a result of the kinetic heating, was considered.

As can be seen according to Fig. 2.16, range with an increase p_0 always grows up to very high values of p_0 ($p_{opt}=900$ kgf/m²). The decrease of range with further increase in p_0 is connected with the incidence/drop in the coefficient ξ.

Note. In the practice of the design of long-range aircraft such high values of p_0 ($p_0=900$ kgf/m²) are not used for following reasons:

1) maximum $L_{max}=f(p_0)$ is very slanting, i.e., L_{max} grows near p_{opt} it is weak;

2) very high values p_0 worsen/impair takeoff and landing characteristics of aircraft.

With increase in p_0 takeoff run length $L_{раз}$ increases.

Examining formula (2.17')

$$L_{раз} \approx \frac{0.82 p}{c_{y\,ot}\,(\bar{P}_0 - f)}\,.$$

it is possible to say following. With increase in p_0 increases that arranged/located P_{cp} as a result of the decrease of that arranged/located $\bar{G}_{K}$. Drag coefficient

$$c_x = k_0 c_{xp} + c_{x\phi} \frac{p_0}{k_1} + D_0\left(\frac{p_0}{q}\right)^2$$

with increase in p_0 increases, or p_0/k_1 - increases, and $p_0 q = c$ remains constant, since the angle of attack of aircraft on the takeoff/run-up depends only on the geometric parameters of chassis/landing gear and remains constant/invariable.

However, comparing formulas (2.17'), (2.26) and (2.25'), it is not difficult to draw conclusion that with increase in p_0 denominator of fraction (2.17') grows more slowly than the numerator and, consequently, $L_{p\cdot c}$ increases with an increase in p_0.

Coefficients ξ and ψ, which characterizing the dependence also of specific consumption of fuel of engine on flight speed, are determined by selected parameters of the engine and virtually, during design of aircraft under existing engine, they are located out of effect of designer of aircraft.

From formulas (2.3); (2.5); (2.6); (2.15') it is evident that increase in coefficient ξ with constant values of other parameters leads to increase M_{max}; $M_{крейс}$; $H_{пот}$; $V_{t\,max}$; L_{max}. However, it follows to have in mind that by the rational value ξ there will be such that with given M_{max} or $M_{крейс}$ will be determined from the formula

$$\xi=\frac{4650 M_{max}^{2} c_{x}}{\overline{P}_{0} p_{0}} \text{ or } \xi=\frac{6950 M_{крейс}^{2} c_{x_0}}{\overline{P}_{0} p_{0}}.$$

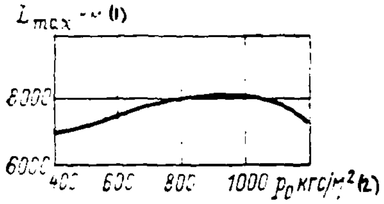

Fig. 2.16. Dependence of maximum range L_{max} on specific load p_0 taking into account effect of p_0 on over-all payload ratio of construction/design

$\overline{G}_{к}$ (p_0=var; $\overline{G}_{т}$=var)

Key: (1). km. (2). kgf/m².

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In this case there must be observed the condition

$$\overline{P}_{0}=\frac{0{,}82 p_{0}+L_{разб} c_{y\,отр} f}{L_{разб} c_{y\,отр} \varepsilon_{ф}}.$$

where $\varepsilon_{ф}=\frac{\overline{P}_{0ф}}{\overline{P}_{0}}$ - coefficient of afterburning;

$\overline{P}_{0ф}$ - thrust-weight ratio on the takeoff with afterburner;

$L_{разб}$ - prescribed/assigned distance of takeoff/run-up;

f - coefficient of friction.

After leading conversions in expressions (2.25) and (2.25'), it is possible to write for prescribed/assigned duration of flight t '

$$4650 M_{max}^{2} c_{x}(r_{0}+S\psi\xi\Delta c_{p0}t')-\xi p_{0}(1-\overline{G}_{к}-\overline{G}_{с.в.т})=0 \qquad (2.31)$$

and for prescribed/assigned range L_{max}

$$6950 M_{крейс}^{3} c_{x_0} r_{0}-M_{крейс}\xi p_{0}(1-\overline{G}_{к}-\overline{G}_{с.в.т}-u)+$$
$$+SRc_{p0}\xi p_{0} L_{max}\sqrt{c_{x_0} D_{0}}=0. \qquad (2.31')$$

Equations (2.31) and (2.31') give demonstrative representation about dependence of most important aircraft performance of M on different parameters and they make it possible to determine the values M_{max} or $M_{крейс}$, which is virtually possible for aircraft in prescribed/assigned parameters $\bar{G}_{кр}$, $\bar{G}_{с.ат}$, p_0, c_{x0}, c_{p0}, r_0, L_{max}, t' and so forth.

Solving first equation (2.31), we will obtain for M_{max} with mission time t '

$$M_{max} = \sqrt{\frac{\xi p_0 (1 - \bar{G}_к - \bar{G}_{с.в.г}}{4650 c_x (r_0 - S - \xi \Delta c_{p} t')}}. \quad (2.32)$$

Note. For determining the value M_{max} it is possible to use this graphic procedure. After assigning several values of M, are determined those corresponding to them ξ, c_x and ψ. The last values we substitute in formula (2.32) and calculate values M_{max}. In terms of the selected values of M and the corresponding to them values M, calculated according to the formula, is plotted a curve in coordinates $M_{зад}$ and $M_{вст}$ (Fig. 2.17), intersection with which with the straight line, carried out from the beginning of coordinates at angle in 45° will give unknown value M_{max}, $M_{зад}$ and $M_{вст}$ (all values they must be undertaken on one scale).

Value $M_{\text{крейс}}$ can be found also, using by graphical solution (Fig. 2.18), after taking as functions for plotting of curves following:

$$\left.\begin{aligned} & 6950 M^3_{\text{крейс}} c_{x_0} r_0 = \theta_1; \\ & M_{\text{крейс}} \xi p_0 (1 - \overline{G}_{\text{к}} - \overline{G}_{\text{с.э.т}} - u) - \\ & - SR \xi L_{max} c_{\mu 0} p_0 \sqrt{c_{x_0} D_0} = \theta_2. \end{aligned}\right\} \quad (2.33)$$

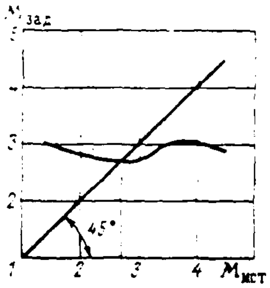

Fig. 2.17.

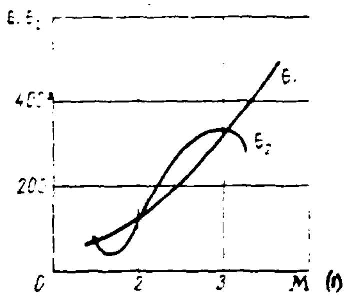

Fig. 2.18.

Fig. 2.17. Example of graphical solution of equations (2.29).

Fig. 2.18. Example of graphical solution of equations (2.29').

Key: (1). M.

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The point of intersection of these curves on the graph in coordinates M and θ_1 and θ_2 will determine unknown value $M_{крейс}$. At some values of quantities, three values $M_{крейс}$ can be obtained In this case to each of them will correspond its value of thrust-weight ratio $\overline{P}_0$, determined from

$$\overline{P}_0 = \frac{6950 M^2_{крейс} c_x}{p_0 \xi}.$$

The presence of several solutions is explained by the complexity of the functions

$$\xi = f(M); \quad \varphi = f(M); \quad c_{x_0} = f(M).$$

Examining equations (2.31) and (2.31'), it is not difficult to draw important conclusion that assigned magnitude M can be obtained only during specific combinations of parameters entering these equations. If the parameters at assigned magnitudes $M_{крейс}$ and L_{max} or t' satisfy equations (2.31) and (2.31'), then the real realization of this aircraft is possible. But if equations are not satisfied, then the designed aircraft in practice cannot be carried out. In addition to this, the named equations reveal/detect direct effect on M of such parameters of aircraft as $G_к, r_с, c_р$ etc. All this speaks about the high value of the equation of the over-all payload ratios of aircraft for the establishment of the regular dependences between the flight characteristics and the most important parameters of aircraft.

Subsequently we will call equations (2.31) and (2.31') fundamental equations of projected/designed aircraft.

On the basis of entire of that presented in this chapter it is possible to do following conclusions.

1. Speeds V_{max} and $V_{крейс}$ (or number M_{max} and $M_{крейс}$) increase, if $\overline{P}_0$ in constant/invariable other parameters increases. The available thrust-weight ratio increases with:

a) the decrease of the specific weight of the power plant r_0;

b) the reduction in the relative weight of the construction/design of aircraft $G_{к}$ (available);

c) the decrease in the specific fuel consumption by engine $c_{р}$ (substantially for the long-range aircraft).

One should, however, remember that maximum speed of aircraft V_{max} is sometimes limited to not available thrust-weight ratio $\bar{P}_0$, but permitted for reasons of strength by velocity head or temperature of kinetic heating.

2. Speeds V_{max} and $V_{крейс}$ can be increased (other conditions being equal) as a result of increase in specific wing load p_0, but in this case it is necessary to consider that with an increase p_0 intensity of increase in velocity V (or Mach number) sufficiently sharply is reduced, and at value of p_0* the increase in V ceases. At the same time with very large p_0 substantially deteriorate takeoff and landing characteristics of aircraft ($V_{пос}$, V_{1}, $L_{разб}$ and $L_{проб}$).

3. Velocities V_{max} and $V_{крейс}$ can be increased with decrease of drag coefficient c_{x0}. The methods of decrease c_{x} are changed in the dependence on rate of speed (subsonic, transonic, supersonic).

4. Static ceiling of aircraft $H_{пот}$ increases ($\Delta_{пот}$ - is reduced), if as a result of measures indicated in p. 1 thrust-weight ratio $\bar{P}_0$ increases.

5. Static ceiling $H_{пот}$ of subsonic aircraft is reduced, if specific load p_0 increases (in this case c_x it increases). For the

supersonic aircraft with the TRD [ТРД - turbojet engine] increase p_0 in certain cases leads to increase $H_{пот}$. Decrease c_{x0} with the constant values of other parameters leads to certain increase $H_{пот}$.

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6. Vertical velocity $V_{y\,max}$ intensely increases with increase in $\overline{P}_0$. Decrease c_{x0} with the constant values of other parameters increases $V_{y\,max}$.

7. Maximum range L_{max} with increase in cruising values $M_{крейс}$ in subsonic range increases [see formula (2.15) and by Fig. 2.7]; with further increase/growth $M_{крейс}$ (with retention/maintaining of other conditions) in transonic range L_{max} is sharply reduced, and with further increase $M_{крейс}$ in supersonic range L_{max} gradually grows. An increase in the specific wing load p_0, which leads to increase $M_{крейс}$, to decrease $\overline{G}_c$ and, consequently, to increase $\overline{G}_т$, in the subsonic and supersonic range (M<1 and M>1.5) to certain value $p_{0\,opt}$ (see Fig. 2.16) leads to increase/growth L_{max}.

8. Distance of takeoff/run-up $L_{разб}$ increases with increase in p_0 and is reduced with increase in $\overline{P}_0$.

9. Fundamental equations of aircraft show that between flight characteristics and most important parameters of aircraft there are laws, by analyzing which it is possible to find rational solutions.

Chapter III.

METHODOLOGY OF THE OPTIMUM DESIGN OF AN AIRCRAFT.

In previous chapter the interdependency of all parameters and characteristics of aircraft, connected with equation of weight balance, is revealed. The task of designer during the development of preliminary design consists of selecting of the optimum combination of the airplane design, power plant, geometric, weight and flight parameters and characteristics.

Thus, contemporary design procedure of aircraft provides for not simply identification of parameters and characteristics on base of statistical data, but selection of optimum parameters, on the basis of single generalized criterion. The methods, with the help of which they solve this problem, are called the methods of optimum design.

Value of optimization of parameters and characteristics of aircraft is difficult to overestimate. Are known the examples, when the non-optimal identification of the parameters of wing forced to reduce by 15-20% payload and thereby to considerably worsen/impair the efficiency of aircraft.

Each percentage of decrease of efficiency/cost-effectiveness of passenger aircraft of average/mean tonnage is equivalent to loss in year of more than 1 mln. rubles in a fleet of one hundred aircraft. Therefore one ought not during the identification of parameters to disregard the possibility of an increase in the efficiency of aircraft even by one percent.

§ 1. Formulation of the problems.

Tasks about identification of parameters and characteristics of aircraft as system of solution (algorithms) of these problems, can be different depending on specific conditions (aircraft type, stage of design, etc.). For example, on the stage of sketch design task consists of the following. With the specific value of payload (commercial or military) for all possible and interesting to the designer airplane designs and types of engines to determine the optimum values of the fundamental parameters and characteristics of the aircraft of such, as takeoff weight, thrust-weight ratio and load on 1 m^2 at the takeoff, the parameters of wing (elongation/aspect ratio, sweepback, relative thickness), the parameters of fuselage and tail assembly, a number of engines and their parameters, flight conditions (velocity, height/altitude, etc.).

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Are considered known general requirements for the comfort of passengers (for commercial airplanes) and flight safety, diagram of takeoff and landings (vertically taking off or with takeoff/run-up), also characteristic of airfield, on which it is proposed to operate aircraft. Problem is solved either with the prescribed/assigned (varied) flying range, or with the prescribed/assigned (varied) flight time.

Such task, in which known combination of values of payload and range (or flight time) is required to determine takeoff weight, engine thrust and other parameters indicated above and characteristic, is called direct problem in contrast to the inverse[1], which is formulated otherwise: is known takeoff weight and calculated flying range (or payload).

FOOTNOTE[1]. Terms "direct" and "inverse" problem make sense in connection with the reversibility of tasks (identical results) with one and the same data. ENDFOOTNOTE.

To find the payload (or flying range, if it is prescribed/assigned is payload), which corresponds to the optimum values of parameters and characteristics of aircraft.

During identification of parameters design of aircraft frequently is conducted under assigned engine. This task in the setting is also reverse/inverse, since the prescribed/assigned takeoff and landing conditions and fundamental flight conditions uniquely determine the takeoff weight of aircraft, and the value, which depends on the optimizable parameters, will be either flying range (with payload $G_{\text{п.н}}=\text{const}$), or payload (when $L_{\text{расч}}=\text{const}$).

The inverse problem is to a known degree is artificial in setting, since in the TTT the value of payload and fundamental flight-performance data usually is indicated. Furthermore, it is desirable to select optimum engine to the aircraft, and not vice versa. Nevertheless task in the reverse/inverse setting they frequently use during calculations, especially when there is no possibility to use computer(s). Matter in the fact that the labor expense for calculations with G_0=const is considerably less than with a change in the takeoff weight. The solution of direct problem is connected with the determination of takeoff weight from the equation of the weight balance of aircraft in each of the versions of calculation. The solution of this equation relatively G_0 succeeds in obtaining only by the labor-consuming method of successive approximations. But if it is used by computer(s), then this difficulty drops out.

From formulations of tasks it is evident that quantity of optimizable parameters and, consequently, depth of optimum they depend on that, how much is superimposed limitations to identification of parameters and characteristics of aircraft. It is obvious that the best results of the solution can be achieved/reached with the smallest number of limitations and the greatest number of optimizable parameters and characteristics.

On the other hand, tendency toward smallest number of limitations during optimization of parameters does not mean that it is possible to solve task without limitations generally. It cannot be forgotten that the tasks in question are engineering, concern completely concrete/specific objects and are designed for the level of the technical capabilities of this time and nearest (visible) prospect for development of technology (metallurgy, technology, the type of power plant).

Special feature of tasks of optimum design is also the fact that optimization of parameters and characteristics of all aircraft components is produced simultaneously. The parameters of parts are connected, and also with parameters and characteristics of entire aircraft equation of the weight balance (see Chapter II). Therefore the optimization of the parameters of separate aggregates/units separately from the optimization of other aggregates/units is the especially approximate procedure.

During solution of extreme problems of design it is important not only to select criterion of optimization (see Chapter 1), but also to agree, what is understood by optimal solution.

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From a purely mathematical point of view the concept of optimum is sufficiently clear - these are point (unit) or locus of points on hypersurface in limited space of n measurement (n - number of optimizable values), where evaluation criterion takes outer limit. However, in the technical engineering tasks this understanding of optimum is narrow and frequently insufficient for the final solution. Everything depends on the character of optimum. If optimum is slanting, then the insignificant digression from it often gives the possibility to obtain different advantages, not placed into the conditions of task (layout, operating, etc.). Therefore by optimal solutions in the technology are understood the solutions, which either correspond to mathematical optimum or they are very close to it, i.e., they do not exceed the limits of the standard deviations (sometimes these solutions are called rational).

§ 2. Methods of the search for optimum. ENDFOOTNOTE.

As a result of high labor expense for calculation exact solution of tasks for simultaneous optimization even of three-four parameters of aircraft is in practice possible only with use by computer(s).

At present known several methods of solution of multiparametric problems of optimization with the help of computer(s).

- classical methods of analysis (methods of extrema or calculus of variations);

- methods of determined search (scanning or gearing/sorting of versions, methods of gradients, fastest descent, etc.) [7];

- method of probabilistic (random) search.

Let us do short survey/coverage of these methods.

Method of extrema, in connection with identification of parameters of aircraft, requires solution of system of equations of form

$$\left.\begin{array}{l} \dfrac{\partial a}{\partial i_1}=0;\ \dfrac{\partial a}{\partial i_2}=0;\ldots\ \dfrac{\partial a}{\partial i_n}=0; \\ k_1 \leqslant k_1(i_1,\ i_2,\ldots,\ i_n); \\ k_2 \leqslant k_2(i_1,\ i_2,\ldots,\ i_n); \\ \ldots\ldots\ldots\ldots\ldots\ldots\ldots \\ \ldots\ldots\ldots\ldots\ldots\ldots\ldots \\ k_m \leqslant k_m(i_1,\ i_2,\ldots,\ i_n). \end{array}\right\} \qquad (3.1)$$

where a - criterion of evaluation of versions;

i - parameter of aircraft;

k - limitation (operating, structural/design, etc.);

n - number of unknown (optimizable) parameters of aircraft;

m - number of limitations.

The method of extrema requires so that all functions, which link criterion of evaluation and their independent variable/alternating, first-order derivatives, and also all functions of limitations of parameters would be continuous in n-dimensional space being investigated, and evaluation criterion must have particular extrema of one form on each of parameters in this space. Only under these conditions there can be obtained general solution of task (for example, the minimum of the prime cost of ton-kilometer). During the use of this method one should, first of all, be convinced (via precomputations and logical considerations) that the required conditions are satisfied.

During solution of system (3.1) it is necessary to follow fact so that none of optimizable parameters would be determined by unambiguously set limitations. If such parameters are, then the derivatives of the criterion of evaluation of aircraft according to these parameters must be, it is understood, they are excluded from system (3.1).

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Classical method of extrema of continuous functions in question possesses number of shortcomings, which limit its practical application for optimization of parameters of aircraft with the help of computer(s). Such shortcomings are: unwieldiness of equations, especially the system of derivatives $\partial a/\partial \lambda_1=0;\ \partial a/\partial \lambda_2=0, \ldots, \partial a/\partial \lambda_n;$ method it does not allow/assume gaps or points of inflection of functions and their derivatives, which forces to reject the optimization by this method of some of the parameters.

Solution by classical method becomes even bulkier, when some of optimizable values are connected with differential equations, for example by equations of motion, it is necessary therefore to use methods of calculus of variations.

If number of optimizable independent parameters is small ($n \leq 6$-8), then most frequently during calculations on computer(s) is used method of gearing/sorting of all permissible versions, known also by the name of method of scanning or blind search. The advantage of the method of scanning in comparison with the classical methods of extrema and extremals consists in the fact that he does not impose special requirements on the form of the function, which link the unknown parameters with the evaluation criterion. Functions and their derivatives must not be continuous, can be assigned them both analytical and tabular methods.

One should stress that scanning is only reliable method of search for global extremum, i.e., most optimum among local extrema.

For this method does not have special importance the character of the extremum: is extremum point (apex/vertex, pocket) or linear (ravine, ridge/spine).

Shortcoming in scanning - large time losses to search, when number of independent optimizable parameters exceeds six to eight.

Time losses to search during scanning can be significantly decreased, if we introduce simple limitation in count: with successive change in parameters are counted only those versions, which lead to favorable change in criterion of evaluation (for example, to decrease of prime cost of ton-kilometer). For decreasing the count time with the gearing/sorting of versions it is possible, furthermore, to reduce a quantity of values of the parameters (to increase the space of search) and it is at first rough, and then ever more accurately to select/take with the help of the computer(s) optimum version, narrowing the region of search. If in this case the edge of n-dimensional cube is divided in half, then this variety of the method of the gearing/sorting of versions can be named the method of the consecutive contraction of the regions of search. In this method is examined the at first wide, but rarefied region, whose boundaries are determined by the extreme values of the parameters, usually known from the structural/design and operating limitations. In this region the machine memorizes the first quasi-optimal version and then it automatically passes to the second cycle of calculation in the region of cube with that reduced doubly edgewise and with the center at the first quasi-optimal point. The process of contracting the regions is conducted until the difference between the latter/last and previous value of the criterion of evaluation of the best versions is exceed the assigned magnitude, for example 0.5%. Latter/last quasi-optimal version is considered optimum. Virtually it is sufficiently two-three contractions of the regions of search.

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Planar diagram of this method is given in Fig. 3.1.

Conversion of values of any of optimizable parameters (x_i) at contraction of region of search can be found from following recursion formula:

$$x_i = x_{i\,opt} \pm \frac{(\Delta x_i)_{z=0}}{2^z}\left[\frac{2\binom{k-1}{N}_0}{k-1} - 1\right], \qquad 3.2.$$

where z=0, 1, 2, ..., m - reference number of cycle of contraction of region;

$\Delta x_{i\,z=0}$ - initial space of change in parameter (with z=0);

$x_{i\,opt}$ - optimum value of parameter at contraction of region (with z=0, 1, 2, ..., m);

N=0, 1, 2, ..., k-1;

k - number of values of parameter at contraction of region of search (usually equal to initial number of values of parameter at z=0).

During use of method of contracting regions of search one should consider that it can be used effectively only then, when there is confidence in existence of only and sufficiently slanting extremum in permissible space. Otherwise the economy of time to the search due to the rarefaction/evacuation of the units of three-dimensional/space grid can lead to the fact that the most optimum version will not be "noticed" by computer(s) with the first circuit/bypass of units (it seemingly it will jump through the sieve with the too great openings/apertures).

Simplicity of research of vicinities of optimum is an essential advantage of method of gearing/sorting of versions. As already mentioned, the not always strict mathematical optimum of the parameters can be realized in practice. Therefore, a change of the parameters in the area of optimum is of interest. With the slanting optimum sometimes there is profitable somewhat to step back from it for any reasons. With the help of the gearing/sorting of versions it is possible to investigate also effect on the criterion of evaluation of the aircraft of each of the independent (optimizable) parameters.

Method of gradient is another spread method of solution with the help of computer(s) of tasks of multiparametric optimization.

Its essence consists of following.

It is known that vector of gradient of scalar function is directed in direction of greatest increase in this function (in this case - evaluation criterion). Therefore optimization according to the method of gradient is reduced to the motion to the side, reverse/inverse gradient. In this case the criterion of optimization will be reduced most strongly. The search is divided off into two stages: the partial derivatives of criterion first are calculated from the optimizable parameters, then system is displaced in the direction, opposite to gradient, is done working space and, etc.

Method of gradient is well adapted to solution of linear tasks, when criterion of optimization is linearly connected with parameters. During the solution the method of the gradient of the nonlinear problems[1], called also problems of nonlinear or dynamic programming, the researcher becomes before this selection: either the direction of search will be not always best, if the space of the search for large or it is necessary to decrease the space of search and to respectively increase the time of search.

FOOTNOTE[1]. This class of tasks includes the optimization of the parameters of aircraft. ENDFOOTNOTE.

Generally gradient method is not characterized by efficiency in time, since before each working step it is necessary to preliminarily analyze the criterion of optimization (to make a large quantity of test spaces).

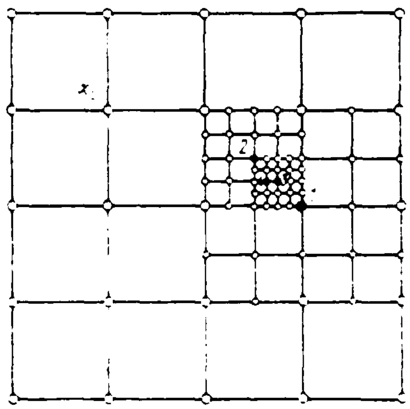

Fig. 3.1. Planar diagram of method of consecutive contraction of regions of search: 1, 2, 3 - quasi-optimal versions; 4 - optimal version during three cycles of contraction; x_i - value of parameter in unit of multidimensional grid.

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For the reduction in time losses to the search, the method of steepest descent is used. From gradient method it differs in terms of the fact that after the first working step (in the most favorable direction) further steps are done in the same direction until the criterion of evaluation of system is changed favorably. In this last unit of multidimensional grid the search for the best direction of further motion according to the method of gradient, etc. is again produced. Thus, the time of search here is reduced due to the decrease of the space of analysis. However, the precision/accuracy of search is not guaranteed, since its direction can be in the side from the global optimum, if the same exists.

Planar diagram of methods of gradient and fastest descent is given in Fig. 3.2.

Recently during selection of optimum parameters of different machines and processes won acceptance method of random (statistical) search, based on probabilistic principle. By propagation this method is obliged to the development of technical cybernetics, to the appearance of high speed computers, without which virtually could not be fulfilled and statistically developed the large number of calculations (random tests/samples).

In methods, described above, either completely is ignored previous experiment of search for optimum or undertakes base previous successful result. In contrast to them, statistical method provides for storage and use of information during the search, this method is adapted to the self-instruction.

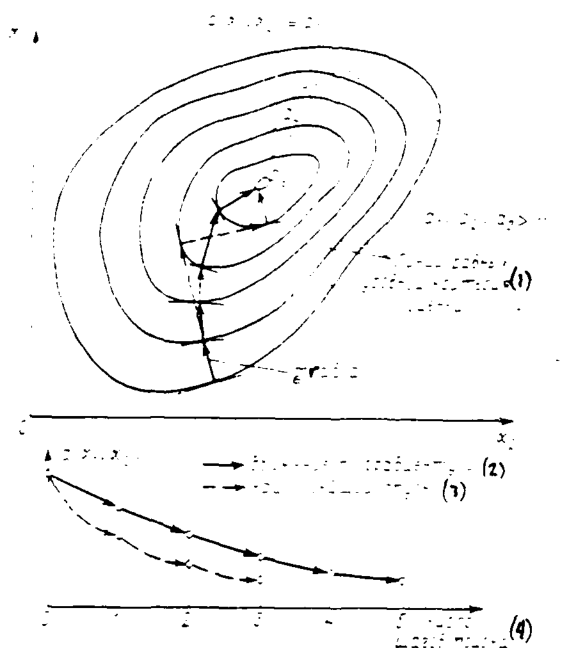

Fig. 3.2. Planar diagram of methods of gradients and fastest descent.

Key: (1). Lines of equal levels of evaluation criterion. (2). motion on gradient. (3). fastest descent. (4). Number of spaces of search.

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As in gradient method, with a search with statistical method calculation begins from randomly selected non-optimal version, which lies, however, in permissible parametric domain. By statistical processing of random tests/samples is done the subsequent step in that direction, where the probability of an improvement in the evaluation criterion is greatest, moreover with an increase in the quantity of spaces grows the confidence of motion, since it is used, as already mentioned, previous experiment of motion to the optimum. Usually it is necessary to do several ten spaces of search, their quantity virtually not depending on a number of optimizable parameters. Especially one should stress that the statistical method gives the large economy of the time of the search for optimum version with a very large quantity of independent parameters (n≥8-10). If researcher decided by simple the countershaft of all possible versions to find with the help of the computer(s) the optimum values of ten parameters at 4-5 values of each of them, then it was necessary to count 4^{10}-5^{10} versions, i.e., from one to ten million. This calculation in the machine, which possesses operating speed in 10-20 thousand operations per second, would occupy ten and hundreds f days of continuous operation, which, of course, is unacceptable. But if we in this case use the method of the random search, where the direction of the subsequent search depends on the results of previous, calculation will occupy a total of several hours.

§ 3. Design and operating limitations.

Besides limitations, placed by assignment ($G_{\text{пн}}$, L, M, etc.) and which are parameters of very problem of optimum design, should be considered also structural/design and operating limitations, placed on optimizable parameters of aircraft.

Let us consider some typical from these limitations.

With variation of takeoff load on 1 m^2 and sweepback of wing of subsonic aircraft one should consider that combination of large load p_0>500 kgf/m^2) and low sweepback ($\chi \leq 20°$) is virtually impossible with H>10000 m from conditions of buffeting and wing heaviness. This is evident from Fig. 3.3, where the values of the permissible (from the conditions of buffeting) and required for the level flight lift coefficients are given. With an increase in the sweep angle of the wing coefficient $(c_y)_{\text{доп}}$, as is known, grows with M=0.6-0.9. From Fig. 3.3 evident also that the aircraft with the unswept wing in flight at height H≥10000 m with M≥0.7 must have $p^{\circ} \leq (400\text{-}450)$ kgf/m^2, which is characteristic to the medium and small aircraft (G_0<80 t).

Let us note that coefficient $(c_y)_{\text{доп}}$ of subsonic aircraft corresponds to beginning of nonlinearity c_y on angle of attack (with increase α) and to beginning of formation/education of "ladle" in curve $m_z(\alpha)$.

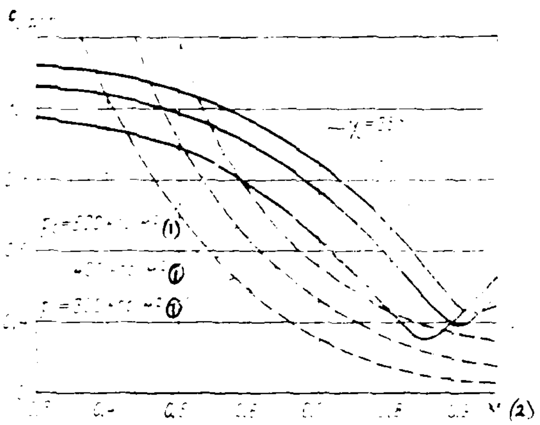

Fig. 3.3. Permissible and required values c_y depending on Mach number of flight (H=10000 m; [illegible] ——— allowed values; - - - - required values.

Key: (1). kgf/m². (2). M.

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It is necessary that with standardized indicator gust of vertical wind $W_i=15$ m/s condition

$$(c_y)_{W_i=15} \leqslant (c_y)_{доп},$$

where

$$c_{y\,W_i=15} = c_{y} - \Delta c_y;$$

$$\Delta c_y = \frac{W}{V} c_y^{\alpha};$$

$$W_i = 15 \frac{}{k} = 15\frac{2 - H}{20 - H} \text{ м/s}$$

would be satisfied Here H in km.

The space required for arranging the fuel is another operational design limitation of parameters of wing. If fuel/propellant must be located only in the wing (civil/civilian or military transport aircraft), then from the parameters of wing, mainly on area and relative thickness, depends its space. Therefore the combination of the parameters, during which the necessary quantity of fuel/propellant is not placed in the wing, is unacceptable. The over-all payload ratio of fuel/propellant, permitted by the space of wing, can be found from the following approximation formula[1]:

$$(\bar{G}_T)_{\text{кр. доп}} \simeq \frac{\beta \bar{c}_0}{p_0} \sqrt{\frac{G_0}{p_0 \lambda}}. \qquad (3.3)$$

FOOTNOTE[1]. More precise formulas are given in Chapter XV. ENDFOOTNOTE.

Here β=265, with the wing taper η=3;

β=280, with the wing taper η=4.

If wing taper in plan/layout is excellent from values (in limits $2.5 \leq \eta \leq 4.5$) indicated, then coefficient β can be found with the help of linear interpolation. These data correspond to subsonic aircraft.

Formula (3.3) it is possible to use, also, during sketch design of the SPS, for which in the first approximation, β=350-370.

During solution of corresponding problem required over-all payload ratio of fuel/propellant is equal with permissible over-all payload ratio of fuel/propellant. If at any version of the combination of parameters of aircraft $(\overline{G}_{т})_{потр} > (\overline{G}_{т})_{кр.доп}$ then these versions are rejected as unrealizable.

On supersonic military aircraft arrangement/position of part of fuel/propellant in fuselage is possible. In these cases formula (3.3) can be used during the calculation of the relative fuel load, which must be placed by f fuselage,

$$(\overline{G}_{т})_{ф} = (\overline{G}_{т})_{потр} - (\overline{G}_{т})_{кр.доп}, \text{ if } (\overline{G}_{т})_{потр} > (\overline{G}_{т})_{кр.доп}.$$

Here $(\overline{G}_{т})_{кр.доп}$ - over-all payload ratio of the fuel/propellant, placed in the wing.

From commercial airplanes it is required so that landing run under all operating conditions with maximum permissible landing weight would be not more than 75% available length of landing strip[2] in cases of landing without use of high-lift device of wing or with high-lift device of wing, but without use of reversal of thrust of engines.

FOOTNOTE[2]. The available length of landing strip is equal to length of runway together with the length of clear zone. ENDFOOTNOTE.

In first case - during landing without use of high-lift device of wing - we have according to [3]

$$p_{c} \leqslant \frac{L_{\text{проб. max}} g \rho_0}{2\bar{G}_{\text{пос. max}}}\left[2c_{y\,\text{пос. б. м}}\left(f_{\text{прив}} + 0.4\frac{n_{\text{дв. рев}}}{n_{\text{дв}}}\bar{P}_0\right) + c_{x\,\text{тр. с}}\right] \text{kg/m}^2 \quad (3.4)$$

where $c_{y\,\text{пос. б. м}}$ - lift coefficient of wing without mechanization/lift-off device.

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In second case - during landing with high-lift device of wing, but without use of thrust reversal - we have:

$$p_c \leqslant \frac{L_{\text{проб. max}} g \rho_0}{2\bar{G}_{\text{пос. max}}}\left[f_{\text{пр}} \, [illegible] \, 2c_{y\,\text{пос. м}} - c_{y\,\text{тр. с}}^{*} - c_{x\,\text{тр. с}}\right] \text{kg/m}^2 \quad 3.5$$

where $c_{y\,\text{пос. м}}$ - lift coefficient of wing with mechanization/lift-off device.

This case, as a rule, gives high values of p_0, i.e., it is less rigid in comparison with first case.

Thus, load on 1 m^2 of the wing has operational design limitations on top - from conditions of buffeting and stalling in flight at high altitude, and also from conditions for landing (among other things of conditions of given speed with landing approach).

Besides those examined, they are possible, it is understood, and other operational design limitations of optimizable parameters, which depend on concrete/specific assignment, diagrams and layouts of aircraft.

§ 4. Algorithms of problems.

Algorithms of optimization[1] of parameters and characteristics of aircraft depend, as already mentioned, from formulation of problems, and also from type of aircraft being designed.

FOOTNOTE [1]. Here we have in mind the specific computational methods. ENDFOOTNOTE.

Here it is not possible to present algorithms for aircraft of all types. Are given below several examples, with the help of which it is not difficult to explain the bases of the compilation of calculating algorithms.

For examples are undertaken land-based aircraft with usual takeoff, which have not less than two engines. It is assumed that the section of the steady (cruising) flight is considerably more than than the sections of the gain of altitude and reduction/descent.

In any methods of search for optimum calculation of version of combination of parameters and characteristics of aircraft (assembly of multidimensional grid) is necessary. By algorithms here should be understood the procedure of calculation of version. As the basis of this methodology the known methods of aerodynamic and weight designs lie/rest. However, each task has its of the special features, which should be considered during the compilation of algorithms.

Let us examine the problem in direct (A, B, C) and inverse (C, D, E) settings.

Direct problem.

In this setting (see § 1) it is prescribed/assigned or is known combination of computed values of payload and flying range.

Following cases are of interest.

A. Cruising speed is function of parameters of wing and entire aircraft, flight altitude - by independent variable, optimized together with other parameters of aircraft;

B. Cruising speed and altitude of flight are known (known their combination);

C. Cruising Mach number of flight is prescribed/assigned, flight altitude is the function of the parameters of wing and entire aircraft.

Cases indicated are encountered during design of aircraft and exhaust majority of versions of direct problem, which are of interest.

Case B is characteristic fact that not flight conditions (V, H) depends on parameters of wing as in the case of A, and vice versa - parameters of wing and entire aircraft are located (they are optimized) depending on combination of rated speed and flight altitude.

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Cases indicated relate to subsonic to aircraft, and case C is characteristic for supersonic aircraft.

Let us consider possible sequence of solution in each of given cases.

As criterion of evaluation of versions let us take value of takeoff weight of aircraft[1].

FOOTNOTE[1]. This criterion is very close to the economic criterion of evaluation of aircraft (prime cost of t-km), since it gives approximately the same values of optimum values, as prime cost of a t-km. ENDFOOTNOTE.

The best version of the combination of the parameters and characteristics will correspond to $(G_0)_{min}$.

Case A. It is necessary to optimize the fundamental parameters of subsonic aircraft with DTRD [turbofan engine]. They are known:

- greatest payload and conditions for its arrangement/position in the fuselage;
- flying range, which corresponds to the greatest payload;
- official load;
- class of runway, which determines accelerate-stop distance, and also the minimally permissible angle of the slope of trajectory during the continued takeoff (with one failed engine) under the reaction conditions ($t_{атм} = +30^\circ$ C, $p_{атм} = 730$ mm Hg);
- character of the coating of runways;
- airplane design;
- altitude-speed characteristics of DTRD (thrust and the specific hourly consumption of fuel/propellant on V and H);
- dependence of cruising Mach number of flight, and also of coefficients c_y and $c_{y\,max}$ in the cruise and during the takeoff on parameters of wing;
- dependence of the weight of aircraft components of their parameters.

Any of known values or dependences (limitations of task) can be varied, for example can be changed class of runway, airplane design, etc. In this case their optimum parameters of aircraft will correspond to each combination of the limitations of task, and at a certain combination of limitations the parameters will be best of all possible versions. Then it is possible to speak about the best combination of limitations and parameters of aircraft, for example the combination of the class of runway, the parameters of wing and entire aircraft.

Order of solution in this case can be following.

1. Are found coefficient c_y and Mach number in the beginning of cruise from solution of system of two equations:

$$M=M(\bar{c}_0, \chi, \lambda, c_y); \quad (3.6)$$

$$M=\frac{b}{a}\sqrt{\frac{2p_0}{\rho c_y}}. \quad (3.7)$$

Equation (3.6) gives dependence M ($M_{крит}$), while equation (3.7) is condition for level flight. Coefficient b≈0.96-0.97 - considers burnout during the set of cruising altitude; the speed of sound a and the mass air density ρ at the given altitude are known; the combination of the optimizable values $\bar{c}_0$, λ, χ, p_0 for this version is also known.

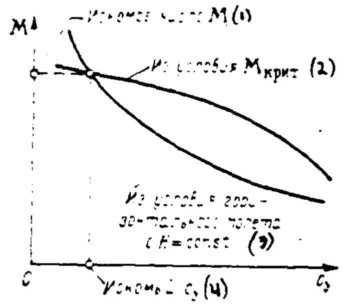

Fig. 3.4. Determination of coefficient c_y and Mach number of flight at base altitude.

Key: (1). Unknown quantity M. (2). From condition. (3). From condition for level flight with H=const. (4). Unknown.

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Determination of Mach number of flight and coefficient c_y is illustrated by Fig. 3.4.

If in specific range c_y $(0 < c_y \leqslant 0.8)$ difference $M_{(3.7)} - M_{(3.6)} > 0$, then these versions of combination (p_0, $\tilde{c}_0$, λ, χ) are excluded, since equations do not have solution. Here $M_{(3.7)}$ and $M_{(3.6)}$ - Mach number from equations (3.7) and (3.6).

2. With known values of H and a and obtaired Mach number cruising flight speed $V_{крейс} = aM$ is determined

3. Is located specific hourly consumption of fuel $c_P = c_P(M, H, m, k_{дг})$, where m - degree of ducted-fan nature of DTRD (optimizable value),

k - throttling coefficient of engines.

4. From solution of system of equations (3.8) values $G_0, c_{x\,m}, \overline{P}_0, K, \overline{G}_т, \overline{G}_{пуст}$

$$\left.\begin{aligned}
&1=\overline{G}_{пуст}(G_0, \lambda, \overline{P}_0, \ldots)-\overline{G}_т(c_p, L_{расч}, c_{x\,m}, \ldots)-\frac{G_{п.н}-G_{служ}}{G_0};\\
&c_{x\,m}=c_{x\,min}(\overline{c}_0, \mathrm{Re}, \overline{S}_{\ldots}, \overline{S}_{\ldots}, \ldots);\\
&\overline{P}_0=\overline{P}_0(L_{пр\,пр}, \theta_{отк}, n_{\ldots}, V_{крейс}, H, c_{x\,min}, \ldots);\\
&K=\frac{c_y}{c_{x\,m}+\frac{(c_y-c_{y0})^2}{\pi\lambda_{эф}}};\\
&\overline{G}_т=\overline{G}_т(c_p, L_{расч}, V_{крейс}, K, m, \ldots);\\
&\overline{G}_{пуст}=\overline{G}_{пуст}(G_0, \lambda, p_0, \overline{c}_0, \chi, \overline{P}_0, \ldots)
\end{aligned}\right\}\quad(3.8)$$

are determined.

The first of equations of system (3.8) is equation of sum of over-all payload ratios of aircraft components. It, as a rule, is not solved explicitly relative to G_0 as a result of complicated dependence $G_{пуст}(G_0)$. Therefore takeoff weight is necessary to find the method of successive approximations.

Minimum value of coefficient of parasite drag of aircraft $c_{x\,min}$ (when $c_y=c_{y0}$) is function of takeoff weight, parameters of wing and flight characteristics, since Reynolds number is equal

$$\mathrm{Re}=\frac{V_{крейс}}{\nu}\sqrt{\frac{S}{\lambda}}=\frac{V_{крейс}}{\nu}\sqrt{\frac{G_0}{p_0\lambda}}.$$

Takeoff thrust-weight ratio $\overline{P}_0$, entering system (3.8), is determined of all interesting designer conditions (interrupted takeoff/run-up, continued takeoff with $\theta=\theta_{отк}$, guarantee of obtained earlier Mach number of flight at altitude H, from condition of ceiling, etc.). Values $\overline{P}_0$, found from these conditions, are equal also as the calculated is taken the greatest value $\overline{P}_0$ with each version of the combination of optimizable values $(H, p_0, \lambda, \chi, \bar{c}_0, m, n_{дв}, \ldots)$.

Lift-drag ratio K, necessary for calculating over-all payload ratio of required fuel/propellant $\overline{G}_т$ [see (3.8)], it is determined from that found earlier system (3.6)-(3.7) to lift coefficient c_y [c_y - is known from analogous polars, $c_{y0} \approx 0{,}1$—$0{,}15$; $\lambda_{эф} = \lambda_{эф}(\lambda, \chi, M, \ldots)$].

Relative empty weight $\overline{G}_{пуст}$, entering system (3.8), is defined as sum of over-all payload ratios of wing, fuselage, tail assembly, chassis/landing gear, equipment and control, power plant according to appropriate weight equations.

In short recording algorithm of task in case in question appears as follows:

$$(M, c_y) \to V_{крейс} \to c_p \to (G_0, c_{x\,\min}, \overline{P}_0, K, \overline{G}_т, \overline{G}_{пуст}) \to \\ \to G_{пуст} \to G_{н.з} \to G_{т\,расх}. \quad (3.9)$$

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In narrow range of change in wing areas (takeoff weights) and flight-performance data of Reynolds number and relative midsections of aircraft components are changed little. If for simplification in the task of disregarding/neglecting the effect of wing area (takeoff weight) on coefficient $c_{x\,min}$, then all values can be determined strictly consecutively/serially. The algorithm of task in this case takes the form

$$\begin{aligned}(M,\ c_y) &\to V_{\text{крейс}} \to c_p \to c_{x\,min} \to \overline{P}_0 \to K \to \\ &\to \overline{G}_т \to G_0 \to G_{пуст} \to G_{к.з} \to G_{т.расх} \ldots \end{aligned} \qquad (3.10)$$

Examples of entire dependences, provided for by algorithms (3.9) and (3.10), are given in Chapter VIII-IX, XV-XVIII.

Case B. In this case of direct problem the height/altitude and the flight speed are known. It is necessary to find the best combination of the parameters of aircraft components $(p_0,\ \lambda,\ \chi,\ \bar{c}_0,\ n_{дв},\ \lambda_ф,\ \ldots)$ at assigned limitations $(G_{пн},\ L_{разб},\ L_{пробег},\ \ldots)$.

Order of solution in this case is by following.

1. For each possible combination H, M, p_0 according to formula (3.7) is determined lift coefficient of aircraft c_y.

2. Using dependence (3.6), with known values of $M,\ c_y(p_0)$ are located combination $\bar{c}_0$, χ, λ, ensuring given Mach number of flight. For example, at first is determined $\bar{c}_0$ with $(p_0,\ \lambda,\ \chi)$=const, then χ with $(p_0,\ \lambda,\ \bar{c}_0)$=const, and so forth. In this case on the parameters of wing $(\bar{c}_0,\ \chi,\ \lambda)$ are placed the limitations:

$$\bar{c}_{0\,min} \leqslant \bar{c}_0 \leqslant \bar{c}_{0\,max};$$

$$\chi_{min} \leqslant \chi \leqslant \chi_{max}; \quad \lambda_{min} \leqslant \lambda \leqslant (\lambda)_{max},$$

on the basis of the structural/design and operating considerations.

The combinations ($\bar{c}_0$, χ, λ), in which are not implemented these limitations, are thrown/rejected.

3. Specific hourly consumption of fuel/propellant in cruise is determined.

4. From system of equations (3.8) $G_0, c_{x\,min}, \bar{P}_0, K, \bar{G}_T, \bar{G}_{пуст}$ are determined. Order of further calculations of the same, as in the case of A.

Thus, in short recording algorithm of task in the case of B appears as follows:

$$c_L \to (\bar{c}_0, \chi, \lambda) \to c_p \to (G_0, c_{x\,min}, \bar{P}_0, K, \bar{G}_T, \bar{G}_{пуст}) \to \to G_{пуст} \to G_{в\,з} \to G_{т\,расх}. \quad (3.11)$$

It is here assumed that $c_{x\,min} = c_{x\,min}(\bar{c}_0, \mathrm{Re}, \bar{S}_{м.ф}, \bar{S}_{сп})$ or, that the same, $c_{x\,min} = c_{x\,min}(\bar{c}_0, G_0, p_0, \lambda, S_{м.ф}, S_{сп})$ with (H, V)=const.

If it is possible to disregard/neglect effect of takeoff weight on coefficient $c_{x\,min}$ (or to average this effect in narrow range of change in Re numbers), then in the case of B it is possible and in (3.10), consecutive determination of all necessary for solution of problem values

$$c_L \to (\bar{c}_0, \chi, \lambda) \to c_p \to c_{x\,min} \to \bar{P}_0 \to K \to \bar{G}_T \to \to G_0 \to G_{пуст} \to G_{в\,з} \to G_{т\,расх}. \quad (3.12)$$

Case C. This case for the subsonic aircraft is examined below, in connection with the inverse problem [when $(G_0, G_{п\,н}) = const;\ L_{расч} = var$].

Let us consider order of calculation in the case of C, in connection with direct problem [when $(G_{г.н.}\ L_{расч}) = \text{const};\ G_0 = \text{var}$] for supersonic passenger aircraft.

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Let us allow, for definition that it is necessary to optimize following independent parameters of SPS:

- average/mean (equivalent) wing chord ratio
- geometric mean sweepback of wing on leading edge $\chi_{п.к}$[1]:
- load on 1 m² of wing during takeoff p_0.

FOOTNOTE[1]. Wing aspect ratio in this case is determined by the law of a change in the sweepback on the leading and trailing edges during the prescribed/assigned spread/scope and the contraction. ENDFOOTNOTE.

Is prescribed/assigned: M number of flight; greatest commercial and official load; range of nonstop flight, which corresponds to $G_{к.н.}$ class of airfield and condition for providing safety of transportation; level of comfort of passengers, sizes of cabin/compartment and seats).

They are known: to dependence of weight of aggregates/units and aerodynamic coefficients on unknown (optimizable) parameters of wing and Mach number; airplane design and sizes/dimensions of fuselage; engine characteristic; service lives and specific costs/values of engines and glider/airframe; composition of crew; number of engine overhauls and glider/airframe in serviceable life of service. In addition to this, are known the structural/design and operating limitations of the values of the unknown parameters of wing of SPS.

As a criterion of evaluation of versions prime cost t-km is accepted or given expend (see Chapter I).

Sequence of calculating version of SPS (during any permissible combination of optimizable parameters) in case in question following.

1. From system of equations (3.13) they find G_0, $c_{x\,\min}$, c_y^{α}, $K_{\max}$, $\bar{S}_{\text{омыв}}$, $\bar{P}_0$, H, $\bar{G}_{\text{т}}$, $\bar{G}_{\text{пуст}}$.

$$\left.\begin{aligned}
&1=\bar{G}_{\text{п.ст}}+\bar{G}_{\text{т}}+(G_{\text{к.у}}+G_{\text{служ}})/G_0;\\
&\bar{G}_{\text{т}}=\bar{G}_{\text{т}}(K,\ c_p,\ L_{\text{расч}},\ M);\\
&K=K(c_{x\,\min},\ c_y^{\alpha},\ c_{yc});\\
&c_p=c_p(M,\ T_з);\\
&c_y^{\alpha}=c_y^{\alpha}(\chi_{\text{п.к}},\ \bar{S}_{\text{омыв}},\ M);\\
&\bar{S}_{\text{омыв}}=\bar{S}_{\text{омыв}}(G_0,\ p_0,\ \chi_{\text{п.к}});\\
&c_{x\,\min}=c_{x\,\min}(\bar{c}_{\text{ср}},\ \chi_{\text{п.к}},\ M,\ \bar{S}_{\text{омыв}},\ \bar{S}_{\text{оп}},\ \bar{S}_{\text{м.ф}},\ G_0,\ p_0);\\
&\bar{G}_{\text{пуст}}=\bar{G}_{\text{пуст}}(G_0,\ \bar{c}_{\text{ср}},\ \chi_{\text{п.к}},\ p_0,\ \bar{P}_0,\ M);\\
&\bar{P}_0=\bar{P}_0(L_{\text{пр.р}},\ L_{\text{пр.н}},\ \theta_{\text{отр}},\ n_{\text{дв}},\ M,\ H,\ c_{x\,\min},\ p_0);\\
&H=H(p_0,\ c_{x\,\min},\ c_y^{\alpha},\ M).
\end{aligned}\right\}\quad(3.13)$$

2. Navigational reserve of fuel

$$G_{\text{к з}} = G_{\text{в з}}[G_0, (K)_{\text{M}-1}, (c_{\text{p}})_{\text{M}-1}, L_{\text{расч}}].$$

is determined.

3. Is located empty weight and expendable fuel/propellant:

$$G_{\text{пуст}} = \overline{G}_{\text{пуст}} G_0; \; G_{\text{т расх}} = \overline{G}_{\text{т}} G_0 - G_{\text{н з}}.$$

4. Scheduled velocity $V_{\text{рейс}}$ is calculated.

5. For this the combination of parameters of wing and limitations of task prime cost of transportation is located. The optimum parameters of aircraft correspond to the minimum of prime cost t-km or to the minimum of the reduced expenditures.

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In short recording algorithm of task for SPS in case C can be you smelled thus:

$$(G_0, c_{x\,\text{min}}, c_y^a, K, \overline{S}_{\text{омыв}}, \overline{P}_0, H, \overline{G}_{\text{т}}, \overline{G}_{\text{пуст}}) \rightarrow \\ \rightarrow G_{\text{к з}} \rightarrow G_{\text{пуст}} \rightarrow G_{\text{т расх}} \rightarrow V_{\text{рейс}} \rightarrow a. \qquad (3.14)$$

Inverse problem.

Let us consider following cases of inverse problem, which are of practical interest:

D. The limitation of cruise setting corresponds to case of A of direct problem [when $(G_0, G_{\text{п н}}) = \text{const}; L = \text{var}$]

E. Limitation of cruise setting corresponds to case A of direct problem [when $(P_0, G_{\text{п н}}) = \text{const}; L = \text{var}$]

Case D. Let us assume that as in direct problem (in the case A), the independent variables (by optimizable parameters) of subsonic aircraft are parameters $p_0, \lambda, \chi, \bar{c}_0, n_{\text{эв}}, m, \lambda_{\Phi}$ and flight altitude H. Cruising Mach number of flight is function p_0, χ, $\bar{c}_0$, λ. Furthermore, $(G_c, G_{\text{пв}}) = \text{const}$ and $L_{\text{расч}}$ is the function of the parameters of aircraft.

Are known the same dependences and limitations, as in direct problem. The sequence of calculating the version of aircraft during any permissible combination of the independent variables in this case is by the following.

1. As in the case of A, from equations (3.6) and (3.7) are located Mach number of flight and coefficient c_y.

2. With known to M, H, a is determined $V_{\text{крейс}}$.

3. Coefficient

$$c_{x\,\min} = c_{x\,\min}(\bar{c}_0, \lambda, \text{Re}, \bar{S}_{\Phi}, \bar{S}_{\text{оп}}),$$

where $\bar{S}_{\text{г}} = \bar{S}_{\text{г}}(\lambda_{\Phi}, \Delta\bar{x}_{\text{т}})$ is located

One should also consider that $\bar{S}_{\text{оп}}$ depends on airplane design.

4. As in all previous cases, is determined starting thrust-weight ratio from several conditions: interrupted takeoff/run-up, continued takeoff, guarantee of cruising speed at height/altitude H, etc. For further calculations value $\bar{P}_0$ greatest of

all conditions is accepted.

5. With known G_0, λ, p_0, χ, $\overline{c}_0$, $\overline{P}_0$ is determined relative empty weight $\overline{G}_{\text{пуст}}$.

6. Having c_y, $c_{x\,\min}$, it is possible to find lift-drag ratio in cruise

$$K=\frac{c_y}{c_{x\,\min}+\frac{(c_y-c_{y0})^2}{\pi\lambda_{\text{эф}}}},$$

where $\lambda_{\text{эф}}=\lambda_{\text{эф}}(\lambda,\chi,M)$.

7. Specific hourly consumption of fuel/propellant in cruise $c_p=c_p(M, H, m)$ is determined

8. From equation of over-all payload ratios of aircraft total over-all payload ratio of fuel/propellant is determined:

$$1=\overline{G}_{\text{пуст}}+\overline{G}_{\text{т}}+G_{\text{п.в}}/G_0+G_{\text{служ}}/G_0,$$

whence $\overline{G}_{\text{т}}=1-\overline{G}_{\text{пуст}}-\overline{G}_{\text{п.в}}-\overline{G}_{\text{служ}}$.

9. Is estimated distance of flight with known fuel reserve.

10. Are determined over-all payload ratios of navigational reserve [illegible] and expendable fuel/propellant $\overline{G}_{\text{т.расх}}$

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11. Having $\overline{G}_{\text{пуст}}$, $\overline{G}_{\text{т.расх}}$, G_0, it is possible to find absolute values [illegible]

If a commercial airplane is projected/designed, then obtained parameters and characteristics are used for determining prime cost of transportation - criterion of evaluation of versions.

In short recording algorithm of task in the case of D appears as follows:

$$(M,\ c_y) \to V_{\text{крейс}} \to c_{x\,\min} \to \overline{P}_0 \to \overline{G}_{\text{п.с}} \to K \to c_p \to$$
$$\to \overline{G}_{\text{т}} \to L \to \overline{G}_{\text{н.з}} \to \overline{G}_{\text{т расх}} \to G_{\text{т расх}} \to G_{\text{п.с}} \qquad (3.15)$$

Case E. In this case $(P_0,\ G_{\text{п.н}}) = \text{const}$. The parameters of wing and flight altitude are the independent variables. Mach number and flying range are the functions of the optimizable independent variables.

Calculation of version in this case can be carried out in this sequence:

1) from equations (3.6) and (3.7) are located Mach number of flight and coefficient c_y;

2) is determined cruising flight speed;

3) from system of equations (3.16) are located $\overline{P}_0,\ c_{x\,\min},\ G_0$:

$$\left.\begin{aligned}
&\overline{P}_0 = \overline{P}_0(L_{\text{прерв}},\ \theta_{\text{отк}},\ n_{\text{дв}},\ V_{\text{крейс}},\ H\ldots);\\
&c_{x\,\min} = c_{x\,\min}(\overline{c}_0,\ \lambda,\ \text{Re},\ \overline{S}_{\text{м.ф}},\ \overline{S}_{\text{оп}});\\
&\text{Re} = \frac{V_{\text{крейс}}}{\nu}\sqrt{\frac{G_0}{p_0\lambda}};\\
&G_0 = \frac{P_0}{\overline{P}_0}.
\end{aligned}\right| \qquad (3.16)$$

Here, as it is earlier, $\overline{P}_0$ is determined from several conditions and for further calculations is taken greatest value $\overline{P}_0$.

Formula $G_0=P_0/\overline{P}_0$ determines maximum permissible takeoff weight[1], which is considered as calculated value.

FOOTNOTE[1]. With an even greater takeoff weight and P_0=const actual starting thrust-weight ratio will be less than permitted. ENDFOOTNOTE.

Sequence of further calculation, beginning from a determination of relative empty weight, does not differ from diagram (3.15).

Thus, algorithm of task in the case of E takes form

$$(M, c_s) \rightarrow V_{\text{рейс}} \rightarrow (\overline{P}_0, c_{x\,\min}, G_0) \rightarrow K \rightarrow c_p \rightarrow \overline{G}_{\text{т}} \rightarrow L \rightarrow \overline{G}_{\text{н з}} \rightarrow \\ \rightarrow \overline{G}_{\text{т расх}} \rightarrow G_{\text{т расх}} \rightarrow G_{\text{пуст}}. \quad (3.17)$$

Analogously it is possible to compose algorithms, also, with other formulations of problems for aircraft of any type.

Examples of different dependences, entering calculating algorithms determination G_0, $\overline{P}_0$, $G_{\text{пуст}}$, $G_{\text{т}}$ and the like), are given in Chapters VIII-XI, XV-XVIII.

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Chapter IV.

APPROXIMATION METHODS OF THE OPTIMIZATION OF THE PARAMETERS OF AIRCRAFT.

In the previous chapter methods of simultaneous optimization of fundamental parameters and characteristics of aircraft, based on use of computer(s), were examined. However, during the design are encountered the cases, when there is not possibilities to use of computer(s) and it is necessary to choose the fundamental parameters of aircraft, using the approximation methods of optimization. Most commonly used of these methods are examined in present chapter.

§ 1. Determination of the particular optima of the parameters of aircraft.

Task is placed as follows. Are known

- the designation/purpose of aircraft and its diagram;
- type and the engine characteristic;
- value of payload and the fundamental flight characteristics, provided for by requirements for the aircraft.

It is necessary to define the particular optima of any interesting designer parameters, for example parameters of wing, fuselage or power plant. During finding of the particular optimum of any of the parameters the remaining parameters are considered known from the statistics or the previous approximate computations and are considered as constant values. This method, called the method of "freezing" of the parameters, is used during optimization of one of them.

If the optimum of any of parameters at given values of payload, range and Mach number of flight is defined, then by approximate criterion of optimization, as it was noted in Chapter I, value of takeoff weight of aircraft can be. In this case the optimum of the parameter will correspond to the minimum of takeoff weight.

When payload and take-off weight (given one or found in the first approximation) are known values, then flying range can be criterion of optimization and optimum of parameters will correspond in this case to maximum of range.

Are possible other more complicated criteria, which consider efficiency/cost-effectiveness of aircraft, for example, prime cost of transportation.

During determination of optima should be considered limitations, placed on parameters and characteristics of aircraft (see Chapter III).

Algorithms, led in Chapter III, especially when is averaged coefficient $c_x(G_0)$, they can be used, it is understood, and for optimization of several parameters of aircraft on base of calculations without use of computer(s).

It is of interest the process of onset of optima of separate parameters of aircraft components.

As examples let us consider onset and approximate graph-analytic determination of particular optima of following parameters of aircraft:

- wing aspect ratios λ;
- wing chord ratio $\bar{c}$;
- sweep angle of wing χ;
- load on 1 m^2 of wing during takeoff p_0;
- elongation/aspect ratio of the fuselage λ_ϕ;
- bypass ratios of TVRD m.

Optimum wing aspect ratio. On wing aspect ratio depend, in essence, two values - weight of wing and fuel loads. With an increase in elongation/aspect ratio ($\lambda \geq 3$) grows the weight of wing in other constant/invariable parameters (including the wing area), since the bending and torsional moments increase, and also the shearing forces. If we as the criterion of optimization take takeoff weight, then an increase λ adversely will affect this criterion (takeoff weight it will increase).

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On the other hand, with increase λ inductive wing drag [17], required thrust of engines and weight of required fuel/propellant will be reduced. The weight of power plant can be approximately considered independent of wing aspect ratio. Thus, the optimum of wing aspect ratio can be approximately determined on the minimum of the sum of the weights of wing and fuel/propellant.

Graphical solution of task about selection λ_{opt} is given in Fig. 4.1 and does not require special explanation.

Obtained optimum λ of wing is, as a rule, moderated on sharpness, and change in evaluation criterion to 1% in comparison with its outer limit gives possibility to step back from λ_{opt} to 8-10%.

Besides graphic, and possible approximate analytical solution, determined from condition of extremum with M≠f(λ)

$$\frac{\partial G_{\kappa p}}{\partial \lambda}+\frac{\partial G_{T}}{\partial \lambda}=0. \qquad (4.1)$$

For example, for the subsonic nonmaneuverable jet aircraft, taking into account satisfaction of condition (4.1), it is possible to obtain the following equation:

$$\lambda_{\text{опт}}^{3}\left[1-(0{,}018-0{,}14\chi^{\text{рад}})\lambda_{\text{р}}\right]=$$

$$=\frac{[illegible](c_{x})_{p}\,[illegible]\,(\bar{c})^{[illegible]}\,(\chi\ \cos\chi)[illegible]}{87G_{0}}\left[\frac{(c_{e})_{cp}\left(L_{[illegible]}-\frac{[illegible]}{[illegible]}L[illegible]\right)}{\varphi\,(V_{\text{крейс}}-50)\frac{[illegible]}{[illegible]}}\right]^{[illegible]}=E. \qquad (4.2)$$

where $\chi^{\text{рад}}$ and χ° - sweepback of wing on 1/4 chords in the radians and in the degrees respectively; $(c_{e})_{cp}=(0.92-0.94)\,c_{e\,\text{крейс}}$ - average/mean for the flight time specific hourly consumption of fuel/propellant in kg/kg·h; φ - coefficient of unloading of wing; b=1.05-1.20 - coefficient, depending on aircraft type (with the decrease of the tonnage of value b they are reduced; approximately $b\approx1.05+G_{0}\cdot10^{-3}$, where G_{0} in t).

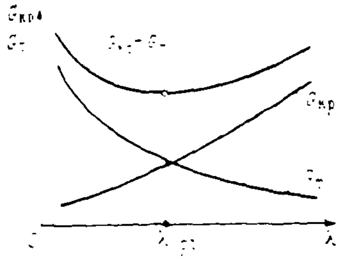

Fig. 4.1.

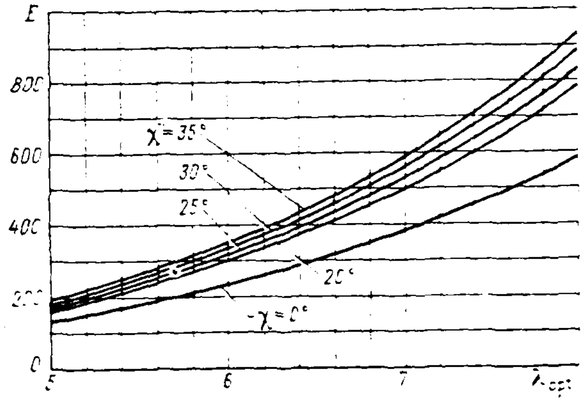

Fig. 4.2.

Fig. 4.1. Schematic of onset of optimum wing aspect ratio.

Fig. 4.2. Graph for determining optimum wing aspect ratio of subsonic nonmaneuverable aircraft.

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Graph for determination λ_{opt} from formula (4.2) is given in Fig. 4.2. The analysis of formula (4.2) makes it possible to do the following conclusions:

- with an increase in the calculated flying range L and specific hourly consumption of fuel/propellant c_p grows optimum wing aspect ratio. For the subsonic aircraft the planned trend in development of TVRD, which foresees reduction/descent c_p implies decrease λ with other constant/invariable parameters of aircraft;

- with increase $c_{x\,n.s}$ in the aircraft (or c_x in the absence of the twist of wing) optimum wing aspect ratio increases;

- with an increase in calculated cruising speed λ_{opt} is reduced;

- increase when L_{pacq}=const takeoff weight (G_0) as a result of increasing the payload reduces optimum wing aspect ratio;

- with an increase in the values of such parameters as p_0, $\bar{c}_0$, η, and also with the decrease of angle χ value λ_{opt} increases. Generally, if a change in any parameter leads to the decrease of the over-all payload ratio of wing, then this change increases λ_{opt} and vice versa;

- greatest effect on λ_{opt} have load on wing p_0 and its sweepback χ.

As show results of calculations, precision/accuracy of determination λ_{opt} from approximation formula (4.2) composes ±10%.

Optimum wing chord ratio. Let us consider how wing chord ratio $\bar{c}$ average/mean on the spread/scope affects the takeoff weight of aircraft with the given values of payload and flying range. We will consider that the law of a change in the wing chord ratios on the spread/scope is known[1].

FOOTNOTE[1]. The solution of the problem about the optimum law of change $\bar{c}$ on the spread/scope of the wing is given in § 3 of this chapters. ENDFOOTNOTE.

Change $\bar{c}_{cp}$ proves to be effect mainly by weight of wing and its aerodynamic drag. With increase $\bar{c}_{cp}$ the weight of the wings of contemporary aircraft in its other constant/invariable parameters is reduced as a result of an increase in the overall height h of wing. With an increase in h with the constant value of bending moment $M_{изг}$ - axial forces $P = M_{изг}/h$ applied to the beam flanges (or to the panels of caisson), are reduced and required cross sections and weight of longitudinal load-bearing elements [2] respectively are reduced.

FOOTNOTE [2]. For some of wing construction increase $\bar{c}_{cp}$ implies an increase in the weight of ribs. However, this fact does not have vital importance, since the over-all payload ratio of the longitudinal structural assembly of contemporary wings is considerably more than the relative weight of ribs. ENDFOOTNOTE.

Thus, increase $\bar{c}_{cp}$ leads to decrease of takeoff weight of aircraft.

On the other hand, with increase $\bar{c}_{cp}$ shaped and wave wing drag grows, which raises required thrust and fuel consumption with given Mach number of flight.

Consequently, with some values of parameters and characteristics of aircraft is possible existence of optimum value $\bar{c}_{cp}$ (Fig. 4.3).

Having dependences $C_{cp}(\bar{c}_{cp})$ and $G_T(P_{n\,cp})$, where $P_{n\,cp}=G\frac{c_y}{c_x}$ and $c_x=f(\bar{c}_{cp})$, it is possible graphically to find $(\bar{c}_{cp})_{opt}$.

Solution of this problem leads to following results ($\bar{c}$ - everywhere along flow):

for supersonic aircraft

$$(\bar{c}_{cp})_{opt}=(2.5-3.5)\%;\ (\bar{c}_0)_{opt}=(3-4)\%;$$

for subsonic aircraft with TVRD

$$(\bar{c}_{cp})_{opt}=(9-12)\%;\ (\bar{c}_0)_{opt}=10-13\%.$$

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Optimum $\bar{c}$ of wing is usually not strong. Change in the evaluation criterion to 1% gives the possibility to step back from $\bar{c}_{opt}$ to 10-12%.

During design of subsonic aircraft determination of $\bar{c}_{opt}$ is somewhat complicated by the fact that flight speed, which corresponds usually M_{cr}, depends on $\bar{c}_{cr}$ or on $\bar{c}_0$. In particular, with increase in c_0 of wing flight speed is necessary to reduce. Therefore, if flight speed enters into the criterion of evaluation of aircraft as, for example, into the prime cost t-km, then during determination of $(\bar{c})_{opt}$ must be also taken into consideration dependence $V_{cr}(\bar{c}_{cr})$.

Optimum sweepback of wing. The sweepback of the wing χ of subsonic aircraft affects, first of all, the weight of wing, fuselage and power plant. An increase of χ with constant/invariable remaining parameters leads to the weight increase of wing due to an increase in the structural/design span (in the direction of 1/4 chord) and the torsional moment in the root cross sections. It is also necessary to reinforce end wing sections for the torsion in order to avoid the reversal of ailerons. Furthermore, if the longitudinal structural assembly of sweptback wing approaches at angle the frames/formers of fuselage, then the part of the bending moment from the wing is transmitted to the fuselage, increasing its weight [1], and is greater, the greater χ.

FOOTNOTE [1]. The weight of fuselage increases also in connection with an increase in the length of its tail section, the caused increase in the sweepback of wing. ENDFOOTNOTE.

Finally, with an increase χ are reduced values $\bar{c}_y^0$ and $\bar{c}_{y\,c\tau p}$ which leads to an increase in the required starting thrust-weight ratio of aircraft.

However, with all enumerated deficiencies increase in sweepback of wing of subsonic aircraft gives important advantage: appears possibility to increase flight speed due to increase in number $M_{\kappa p}$ and reduction/descent c_x, which favorably affects efficiency/cost-effectiveness of aircraft. As if counterweight to the unfavorable effect of an increase in the sweepback of wing by the weight characteristics of aircraft appears. Therefore are possible the existence of the optimum sweepback of wing on 1/4 chords, the corresponding to the minimum prime costs t-km (Fig. 4.4).

If we solve problem about $(\chi)_{opt}$ graphically, taking into account entire contradictory dependences, it is possible to obtain approximately following values $(\chi)_{opt}$ for subsonic jet aircraft:

aircraft of low and medium distance (≤1500 km) - 20-25°;

aircraft of medium distance (2000-3000 km) - 30-35°;

aircraft of long range (≥5000 km) - 35-37°.

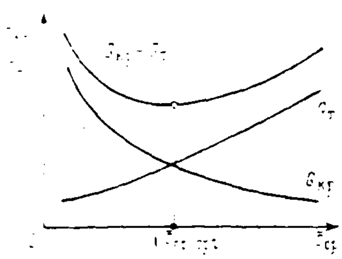

Fig. 4.3.

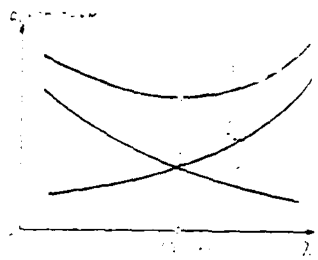

Fig. 4.4.

Fig. 4.3. Schematic of emergence of optimum average/mean wing chord ratio.

Fig. 4.4. Schematic of emergence of optimum sweepback of wing: 1 - effect of flight speed; 2 - gravity effect of construction/design and power plant; 3 - total effect on prime cost of transportation.

Key: (1). a, kopecks/ton-kilometer.

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One should consider that optimum values of sweepback depend also on other parameters of wing, mainly, from $\bar{c}_0$, λ, p_0. With increase in p_0 and $\bar{c}_0$ values $(\chi)_{opt}$ grow, while with an increase λ - they fall. Respectively are chosen values $(\chi)_{opt}$ in the recommended limits.

As far as supersonic aircraft with low-aspect-ratio wing are concerned, effect of sweepback on leading edge χ_{le} by weight of wing and fuselage more complicated than in subsonic aircraft (sweepback of wing of supersonic aircraft it is measured on leading edge). Increase χ_{le} in the delta wing, for example, does not always lead to a gain in weight of wing and fuselage, but beginning from the specific sweepback it leads even to reduction in the weight of these aggregates/units. Value $\chi_{\text{ок}}$ has a stronger effect on lift-drag ratio and fuel load. Approximately it is possible to consider that the optimum sweepback of wing on $M_{\text{1,сq}}$ corresponds to the maximum of lift-drag ratio and is equal (see Table).

[illegible]	2,0	2,3	2,5	2,7	3,0
[illegible]	[illegible]	67	69	72	75

If sweepback along the leading edge is changed on the span, then given data should be accepted as the average values.

Optimum of sweepback of wing is, as a rule, very strong. Change in the evaluation criterion to 1% gives the possibility to step back from $\chi_{\text{ок}}$ in all on 3-4°.

Optimum load on 1 m² of wing during takeoff. Load on 1 m² of wing during takeoff ($p_0=G_0/S$) most strongly affects the weight of wing and power plant, and also weight of fuel/propellant. Schematically this effect is shown on Fig. 4.5.

With increase in p_0 is reduced wing area, which leads in other constant/invariable parameters to reduction in its weight. At the same time with increase in p_0 it is necessary to increase the starting thrust-weight ratio of aircraft ($\bar{P}_0$) in order to satisfy requirements along the length of run-up or length of runway (accelerate-stop distance). Increase in $\bar{P}_0$ leads to the weight increase of power plant.

Effect of p_0 on weight of fuel/propellant more complicated, but relatively weaker. Occurs optimum p_0 by the fuel load, which corresponds to the minimum of its expenditure (point 1 for Fig. 4.5).

Thus, contradictory effect of p_0 on $G_{кр}$, $G_{c.y}$ and G_T leads to existence of optimum of load on 1 m^2 of wing.

Is possible analytical determination $(p_0)_{opt}$ in first and second approximations/approaches.

In the first approximation, rational value of load on 1 m^2 can be obtained from considerations of similarity in dependence on takeoff weight. Actually, wing area is proportional to the square of the linear dimensions

$$S=al^2,$$

and takeoff weight in the first approximation can be considered proportional to the cube of the sizes/dimensions:

$$G_0 = bl^3, \quad (4.3)$$

where (a, b) =const.

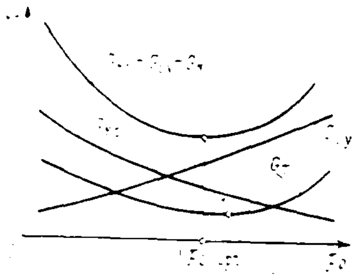

Fig. 4.5. Diagram of emergence of optimum load on m² of wing: 1 - optimum according to per-kilometer expenditure/consumption.

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Consequently,

$$p_0=G_0/S=cl. \qquad (4.4)$$

Substituting from (4.3) $l=\sqrt{G_0/b}$ in (4.4), we will obtain

$$p_0=\xi G_0^{1/3}, \qquad (4.5)$$

where ξ=const.

For subsonic nonmaneuverable aircraft coefficient $\xi \approx 10$, if G_0 in kg, and p_0 in kgf/m² (Fig. 4.6).

From graph Fig. 4.6 it is evident that curve $p_0=10\ G_0^{1/3}$ very satisfactorily maps statistics p_0 (G_0). However, it does not explain the scatter of values p_0 with G_0=const. For example, with G_0=43-44 t there is a range p_0=315-505 kgf/m². The scatter of values p_0 with G_0=const can be explained, if we take into account the dependence of load on 1 m² of wing on the conditions for takeoff and landing - assigned length of runway, c_L with the breakaway and during the landing. With G_0=const there are aircraft, designed on the runways of different length, different speed with the landing approach and different degree of the high-lift device of wing ($c_{y\,отр}$ or $c_{y\,пос}$).

Dependence p_0 (L[illegible]) of transport aircraft is described, for example, by following equation, obtained from coincidence of conditions of interrupted and continued takeoff with assigned climb angle ($\sin\theta$[illegible]) in the case of failure of one engine [3]:

$$[illegible] = k_2 [illegible] L [illegible] + L_{кпт} \left[\frac{1}{k_1} + \frac{1}{K_{отр}} - \sin\theta [illegible] - \frac{f_{каг} - \mu}{k_1}\right] \text{ kgf/m}^2 \quad (4.6)$$

where $k_1 \approx -0.01$ m;

$k_2 \approx 0.83 - 0.017\, m$;

m - bypass ratio of TVRD;

$K_{отр}$ - lift-drag ratio with breakaway;

g=9.81; ρ_0=0.114 with $t_{аэр} = -30$ C, $p_{аэр} = 730$ mm Hg;

$L_{кпт}$ - length of clear zone (250-400 m);

$f_{каг}$ - rolling friction coefficient of wheels;

μ - average/mean air resistance in section of final takeoff, in reference to takeoff weight ($\mu \approx 0.03$-0.05).

Values $\sin\theta_{\text{отр}}$ depend on number of engines and are given by technical requirements (see Table).

$n_{\text{дв}}$	2	3	4
$\sin\theta_{\text{отр}}$	0,025	0,027	0,030

If takeoff run length with MSA is design conditions, then formula for determining load on 1 m² of wing takes form

$$p_0 = g\rho c_{y\,\text{отр}} L_{\text{разб}} (k_3 \overline{P}_0 - f_{\text{кач}} - \mu) \quad \text{kgf/m}^2, \qquad (4.7)$$

where $k_3 \approx 0.92$-0.95.

From condition of given speed with landing approach or landing speed

$$p_0 = c_y \frac{\rho_0 V^2}{2k} \quad \text{kgf/m}^2. \qquad (4.8)$$

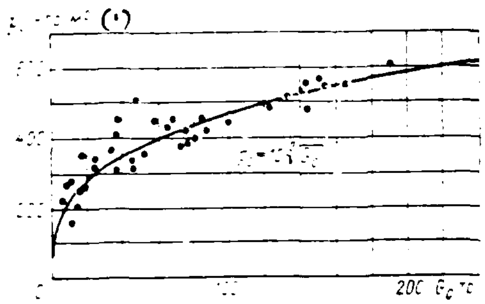

Fig. 4.6. Change of load on m² of wing in dependence on takeoff weight of subsonic aircraft.

Key: (1). kgf/m².

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Here and V are taken with the landing approach or during the landing; k_4 - coefficient, which considers the decrease of landing weight in comparison with the takeoff ($k_4 \leq 1$).

Formulas (4.6)-(4.8) can be used for determining optimum (rational) load on 1 m² of wing in second approximation/approach.

In third approximation/approach $(p_0)_{opt}$ it is determined taking into account effect of p_0 by weight of wing, power plant and fuel/propellant (see Fig. 4.5) and differs from calculation of second approximation/approach insignificantly (to 3-5%), parameters, whose increase leads to gain in weight of wing (G_0, χ, λ), causing also increase $(p_0)_{opt}$ and, vice versa, parameters, whose increase leads to reduction of weight of wing ($\bar{c}_0$, η), they contribute to incidence/drop in value $(p_0)_{opt}$.

Optimum of load on 1 m² of wing is, as a rule, strong. Change in the evaluation criterion to 1% of the outer limit gives the possibility to step back from $(p_0)_{opt}$ in all to 4-6%.

Optimum fineness ratio of fuselage. Fineness ratio of fuselage λ_ϕ the ratio of the length of fuselage to the diameter on the midsection [1], affects, first of all, weight of fuselage itself, and also weight of chassis/landing gear, tail assembly and fuel/propellant.

FOOTNOTE [1]. In the case of noncircular cross section the diameter of a circle, equivalent to the area of midsection, is accepted. ENDFOOTNOTE.

The discrepancy of effect λ_ϕ by weight of these components leads to formation $(\lambda_\phi)_{opt}$ (Fig. 4.7).

Graphical solution of problem about $(\lambda_\phi)_{opt}$ is conducted either at a constant volume of fuselage v_ϕ, when, for example, is known quantity of fuel/propellant, which must be placed in fuselage or in constant floor space $S_{пол}$ - in the case of transport or commercial airplanes, when overall sizes are known and composition of loads or number of passengers and condition for their arrangement/position (level of comfort). In each of these cases the diameter of fuselage, entering the weight formulas, is calculated differently:

from the condition $v_{\text{ф}}=\text{const}$

$$D_{\text{ф}}=\sqrt[3]{\frac{v_{\text{ф}}}{a\lambda_{\text{ф}}}}; \qquad (4.9)$$

from the condition $S_{\text{пов}}=\text{const}$

$$D_{\text{ф}}=\sqrt{\frac{S_{\text{пов}}}{b\lambda_{\text{ф}}}}. \qquad (4.10)$$

Here (a, b)=const - coefficient.

Formulas for determining over-all payload ratio of fuselage and other aggregates/units, necessary for graphic determination $(\lambda_{\text{ф}})_{\text{opt}}$, are given in Chapter XVI-XVIII.

In the first approximation, it is possible to use also formulas for $\bar{G}_{\text{ф}}(\lambda_{\text{ф}})$ and $\bar{G}_{\text{сп}}(\lambda_{\text{ф}})$, given below.

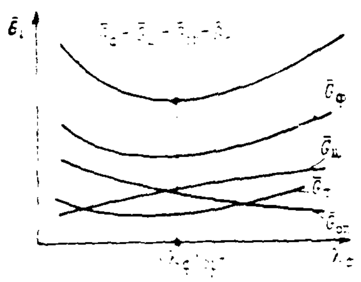

Fig. 4.7. Diagram of formation/education of optimum fineness ratio of fuselage in assigned volume or assigned floor space.

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Over-all payload ratio of chassis/landing gear is connected with fineness ratio of fuselage with dependence of height/altitude of struts on $\lambda_ф$ (Fig. 4.8)

$$\overline{G}_{ш} \simeq c_1 - c_2\lambda_ф - c_3\lambda_ф^2 \qquad (4.11)$$

where c_1, c_2, c_3 - coefficients, which depend on designation/purpose and airplane design.

For example, for civil aircraft:

c_0=0.024-0.026; c_2=0.0018-0.0019; c_3=$3.5\cdot10^{-5}$-$3.6\cdot10^{-5}$.

Over-all payload ratio of tail assembly is function of fineness ratio of fuselage in connection with the fact that area of tail assembly is proportional to length of fuselage

$$\overline{G}_{г} \approx \frac{g_0}{p_0}\,\frac{c_4}{D_ф\lambda_ф}, \qquad (4.12)$$

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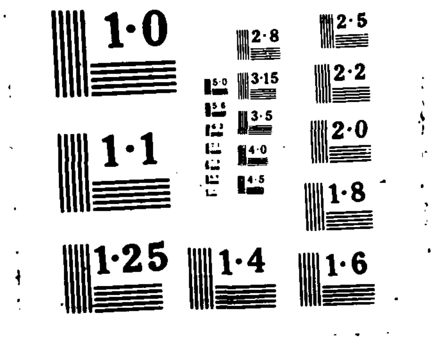
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where $g_{оп}$ - weight 1 m² of tail assembly in kgf/m²;

p_0 - load on 1 m² of wing;

c_4=10-15 - for transport aircraft.

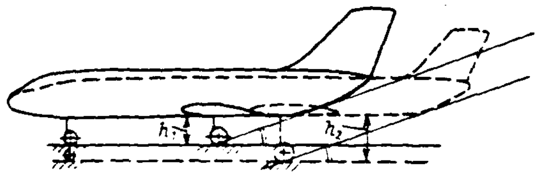
Over-all payload ratio of fuel/propellant also depends on fineness ratio of fuselage (see Fig. 4.7) as a result of dependence of aerodynamic drag of fuselage and engine thrust on λ_ϕ. In the subsonic zone, for example, the coefficient of aerodynamic drag of fuselage in the first approximation, can be calculated from the following formula:

$$c_{x\phi} \approx 0{,}008\lambda_\phi + \frac{0{,}5}{\lambda_\phi^2}. \qquad (4.13)$$

Research shows that optimum of fineness ratio of fuselage is not too sharp/acute. If we consider a change in the evaluation criterion to 1% of the outer limit as permitted, then the possibility to step back from $(\lambda_\phi)_{opt}$ appears to 1-1.5 (to 10-15%).

Optimum bypass ratio of TVRD. The bypass ratio of turbofan engines m, equal to the ratio of the air flow rate through the fan to the air flow rate through the gas-producing part, affects, first of all, the specific hourly consumption and fuel load, by the aerodynamic pod drag (as a result - also by weight of fuel/propellant) and the weight of TVRD themselves. Schematically this effect with M<1 is shown on Fig. 4.9.

In the first approximation, it is possible to consider that for subsonic aircraft $m_{opt}=4-6$, moreover with increase in flying range (or over-all payload ratio of fuel/propellant) value m_{opt} grows.



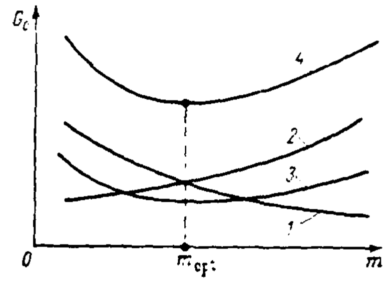


Fig. 4.8.

Fig. 4.9.

Fig. 4.8. Dependence of height of landing gear on length (elongation/aspect ratio) of fuselage ($h_2>h_1$).

Fig. 4.9. Dependence of takeoff weight on bypass ratio of TVRD: 1 - with change in specific hourly consumption of fuel/propellant; 2 - with change in pod drag; 3 - with change in weight of engines; 4 - total change.

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§2. Consecutive optimization of several parameters.

With the help of computer(s) it is possible to carry out simultaneous optimization of large quantity of parameters.

Without application of computer(s) consecutive approximate optimization only of several parameters of aircraft is possible. For this purpose the particular optimization of the parameters, examined in the previous paragraph, is used.

At first is found particular optimum of any from parameters at other "frozen" parameters, known from statistics or from approximate computations. Then the second of the assigned parameters is optimized. In this case the obtained earlier optimum value of the first parameter is accepted. The optimization of the third parameter is conducted with the values of the particular optima, found earlier. The optima of the remaining parameters and characteristics are found thus.

This method of consecutive optimization is, unconditionally, approximated, since after particular optimization of each of subsequent parameters optima of previous parameters must somewhat be changed (on the basis of solution of equation of weight balance, which it connects all parameters and characteristics of aircraft). Therefore after the first cycle of calculations according to the optimization of the parameters it follows, generally speaking, to calculate the second approximation/approach, during which are accepted the values of the parameters, obtained in the first approximation, (in the first cycle of calculations). It is usually completely sufficiently two-three cycles of calculations so that the optima of the parameters would be stabilized at the specific level of values.

§3. Variational problems. Optimum distribution of wing chord ratios on the spread/scope.

During solution of series of problems, when it is necessary to find not only outer limit of parameter, but optimum law of its change (i.e. to optimize functional), are used methods of calculus of variations. As a classical example of the use of the calculus of variations serves the problem about the optimum trajectory, for example about the law of a change in the speed of aircraft on the height/altitude during the lift or the reduction/descent. The optimum law V (H) corresponds to the extremum of any functional - the fuel load or duration of ascent to the base altitude, etc.

Determination of optimum law of bending (twist) of median surface of wing of supersonic aircraft for obtaining maximum lift-drag ratio serves as another example of use of apparatus for calculus of variations during design of aircraft.

In many of these problems in final form it is possible to obtain with specific assumptions of expression for optimum laws of change in functions.

Let us consider about optimum law of change in wing chord ratios on spread/scope as example of use of classical method of variation problem calculus. As the subject of research let us take subsonic aircraft with the trapezoidal ($\chi \geq 0°$) wing.

Problem is placed as follows.

In some known parameters of wing and aircraft (G_0, S, λ, η, D_ϕ) and during assigned flight conditions (V, H, L) to find optimum law of change in wing chord ratios on spread/scope, i.e., to find $[\bar{c}_z(z)]_{opt}$.

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Criterion of evaluation of laws $\bar{c}_z(z)$ with (V, H, L) = const will be incremental value in takeoff weight. Optimum law $\bar{c}_z(z)$ corresponds to the minimum of an increment in the takeoff weight or to the minimum of quite takeoff weight under given conditions. The variation character of the problem in question escapes/ensues from physical considerations.

Actually, if, for example, relative thickness $\bar{c}_z$ remains constant on spread/scope [with (S, λ, η)=const], then weight of wing will be less than in the case, when $\bar{c}_z$ is reduced from one root to the next with $\bar{c}_0$=const. However, aerodynamic drag of "thick" wing with $\bar{c}_z=\bar{c}_0=const$ more than wing drag with decreasing value $\bar{c}_z$ on the span. It is known also that on the value of aerodynamic drag the takeoff weight of aircraft depends, i.e., there is a completely specific weight equivalent of the resistance (see Chapter V), the change in aerodynamic drag considerably more strongly affecting G_0

than the same in the value change in the weight of wing. From these considerations it is evident that version $\bar{c}_2(z)=\text{const}$, will not be best.

On the other hand, if we intensely reduce wing thickness on span, then aerodynamic drag decreases, and weight of wing, vice versa, will be increased with $\bar{c}_0$=const. Obviously, must exist the optimum law of change $\bar{c}_2(z)$, under which the takeoff weight will be minimum.

Let us compose functional of problem, on the basis of calculation of wing to bending and calculation of its profile drag when $M \leqslant M_{\text{кр}}$.

We consider air loads proportional to chords. As a result of the smallness of the over-all payload ratios of the spar webs (conditional beam) and ribs, and also due to the smallness of the effect of dependence $\bar{c}_2(z)$ by weight of walls and ribs during the solution of problem we will not consider their weight. We consider also that the material, working in the general/common bending of wing, together with the walls forms the caisson, sufficient for the perception of the torsional moments.

Thus, during compilation of functional we will consider only weight of material, working in general/common bending of wing, and also component of aerodynamic drag, which depends on $\bar{c}$, profile drag (when $M \leqslant M_{крит}$) The calculation of gross weight of wing or general/common aerodynamic drag (taking into account inductive) in this case is not necessary.

Let us use the connection between change in weight and aerodynamic wing drag, on one hand, and change in takeoff weight - on the other:

$$\Delta G_0 \approx \Delta G_{кр\,изг} \frac{\partial G_0}{\partial G_{кр\,изг}} - \Delta X_{кр\,проф} \frac{\partial G_0}{\partial X_{кр\,проф}} = \Delta G_{кр\,изг} \varkappa_G + \Delta X_{кр\,проф} \varkappa_{X_0},$$

where ΔG_0 - change in takeoff weight of aircraft;

$\varkappa_G$ - derivative of takeoff weight in connection with change in weight of aircraft components (see Chapter V), in this case - weight of wing $(\varkappa_G = \partial G_0 / \partial G_i \approx \partial G_0 / \partial G_{кр})$;

$\varkappa_{X_0}$ - derivative of takeoff weight in connection with change in parasitic aerodynamic drag of aircraft components (see Chapter V), in this case - profile drag of wing $(\varkappa_{X_0} = \partial G_0 / \partial X_{кр\,проф})$.

$$\Delta G_{кр\,изг} = G_{кр\,изг\,0} - G_{кр\,изг\,i};\quad \Delta X_{кр.проф} = X_{кр.проф\,0} - X_{кр.проф\,i};$$

$G_{кр\,изг\,0}$ and $X_{кр\,проф\,0}$ - weight of the material, working in the bending of wing, and the profile drag of initial wing respectively;

$G_{кр\,изг\,i}$ and $X_{кр\,проф\,i}$ - the same for the wing, the differing from initial by dependence $\bar{c}(z)$.

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In this case

$$G_{кр\,изг} + X_{кр\,проф} = \text{const}$$

Let us designate

$$\varkappa_G G_{\text{кр.изг }0}=N_1=\text{const};\quad \varkappa_{X_0} X_{\text{кр.проф }0}=N_2=\text{const}.$$

Then we have

$$-\Delta G_0=\varkappa_G G_{\text{кр.изг }i}-\varkappa_{X_0} X_{\text{кр.проф }i}-N_1-N_2. \qquad (4.14)$$

Weight of shelves of longerons/spars (panels), which receive bending of wing, taking into account discharging will be [22]

$$G_{\text{кр.изг }i}=\frac{4\gamma\varphi}{\xi f \sigma_{\text{ср}}}\int_0^{\frac{l-L_G}{2}}\frac{M_z}{c_z}\,dz. \qquad (4.15)$$

where c_z - greatest profile thickness of wing in cross section z;

γ - specific weight/gravity of material;

φ - coefficient of discharging;

ξ - coefficient, which considers effective height/altitude of shelves (panels) in comparison with maximum;

f - coefficient, which considers decrease of medium stress/voltage in comparison with maximum;

$\sigma_{\text{ср}}$ - medium stress/voltage in upper and lower shelf (panel), determined from relationship/ratio

$$\frac{1}{\sigma_{\text{ср}}}=\frac{1}{2}\left(\frac{1}{\sigma_{\text{в}}}+\frac{1}{\sigma_{\text{сж}}}\right).$$

Here $\sigma_{\text{в}}$ - ultimate tension;

$\sigma_{\text{сж}}$ - allowable stress during the compression.

In formula (4.15) M_z - the bending moment in the current cross section z:

$$M_z = \frac{G_c n_y}{3} \frac{1}{l} \frac{2b_k - b_z}{b_0 - b_k} z^2. \qquad (4.16)$$

where b_k - end wing chord;

b_0 - wing chord along the axis of aircraft;

b_z - current wing chord;

z - coordinate on the spread/scope from the wing tip.

After substitution M_z (4.16) into formula (4.15) and simple conversions from condition $\sigma_{cr}(z) = \text{const}$ we will obtain

$$G_{cr.p.w.} = \frac{4 n_y \gamma \varphi G_c}{3 \xi f \sigma_{cr} l (\eta - 1)} \left[2 \int_0^{\frac{l - D_f}{2}} \frac{z^2 dz}{\bar{c}_z b_z} + \int_0^{\frac{l - l_f}{2}} \frac{z^2 dz}{\bar{c}_z b_k} \right]. \qquad (4.17)$$

Here $\bar{c}_z = \frac{c_z}{b_z}$; $\eta = \frac{b_0}{b_k}$.

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Let us produce in (4.17) replacement of variables, after designating $z' = dz/d\bar{c}_z$. Then

$$G_{cr.p.w.} = N_3 \int_{\bar{c}_k}^{\bar{c}_0} \left(\frac{2z^2 z'}{\bar{c}_z b_z} + \frac{1}{b_k} \frac{z^2 z'}{\bar{c}_z} \right) d\bar{c}_z. \qquad (4.18)$$

where $\bar{c}_0$ - wing chord ratio along side the fuselage [1] without taking into account overflows;

$$N_\xi = \frac{4 l \gamma \psi G_0}{3 \xi f \sigma_{cp} (\gamma - 1)} = \text{const.}$$

FOOTNOTE [1]. It is assumed that to the side of fuselage from the axis of aircraft the wing chord ratio does not vary. ENDFOOTNOTE.

Here γ is accepted in kg/m³; G_0 - in kg; σ_{cp} - in kg /m²; l m; ξ=0.85-0.90; f=0.80-0.85.

Variable part of aerodynamic wing drag $X_{кр.проф}$ entering equation (4.14), can be found as follows:

$$X_{кр.проф} = c_{xp} S q. \tag{4.19}$$

For determination coefficient of profile drag of wing when $M \leq M_{крит}$ we will use known formula of A. A. Dorodnitsyn - L. G. Loytsyanskiy [15]

$$c_{xp} = 2 c_f \eta_c n_1, \tag{4.20}$$

where c_f - coefficient of friction of plate;

n_1 - empirical coefficient ($n_1 \approx 1.485$);

$$\eta_c = \eta_c(\bar{c}, \bar{x}_t);$$

$\bar{x}_t$ - relative coordinate of transition/transfer of laminar boundary layer into turbulent (in chord from nose/leading edge).

For the values $\bar{x}_t = 0.25-0.30$

$$\eta_c \approx 1 + 3.9\bar{c}.$$

Value $2c_f$ we take according to Schlichting [15]:

$$2c_f = \frac{0.91}{(\lg Re)^{2.58}}.$$

Substituting the value of coefficients into formula (4.20), we will obtain for wing with $\bar{c}_z = \text{const}$

$$c_{xp} = \frac{1.35(1 - 3.3\bar{c}_z)}{(\lg Re)^{2.58}}.$$

If $\bar{c}_z = \bar{c}_{z\,cp}$, then

$$c_{xp} = \frac{1.35[1 - 3.3(\bar{c}_z)_{cp}]}{(\lg Re)^{2.58}}, \tag{4.21}$$

where average/mean value $\bar{c}_z$ is equal

$$(\bar{c}_z)_{cp} = \frac{2}{l - D_\phi} \int_0^{\frac{l-D_\phi}{2}} \bar{c}_z dz. \tag{4.22}$$

Substituting (4.21) and (4.22) in (4.19), after conversions we will obtain

$$X_{\kappa\,\text{ckp}} = Sq\left[\frac{8.9}{(\lg Re)^{2.58}(l - D_\phi)} \int_{\bar{c}_\kappa}^{\bar{c}_0} \bar{c}_z z' d\bar{c}_z - \frac{1.35}{(\lg Re)^{2.58}}\right]. \tag{4.23}$$

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Formula (4.14) taking into account (4.18) and (4.23) will take form

$$-\Delta G_\kappa = \gamma_G X_\kappa \int_{\bar{c}_\kappa}^{\bar{c}_0} \left(\frac{2z'z'}{\bar{c}_z t_z} + \frac{1}{t_\kappa}\frac{z'z}{\bar{c}_z}\right) d\bar{c}_z -$$

$$-x_{X_0} N_4 \int_{\bar{c}_k}^{\bar{c}_z} \bar{c}_z z' d\bar{c}_z + x_{X_0} N_5 - N_1 - N_2. \quad (4.24)$$

where $N_4 = \frac{8,9Sq}{(l - D_\ddagger)(\lg Re)^{2,58}} = \text{const};$

$$N_5 = \frac{1,35Sq}{(\lg Re)^{2,58}} = \text{const}.$$

It is joined first and second term in (4.24)

$$-\Delta G_2 = \int_{\bar{c}_k}^{\bar{c}_z} \left[x_G N_3 \left(\frac{2z^2 z'}{\bar{c}_z b_z} - \frac{1}{b_K} \frac{z^2 z'}{\bar{c}_z} \right) + x_{X_0} N_4 \bar{c}_z z' \right] d\bar{c}_z + x_{X_0} N_5 - N_1 - N_2. \quad (4.25)$$

Here $(x_{X_0}, N_5, N_1, N_2) = \text{const}.$

Finally functional of this problem takes form

$$F = \frac{2x_G N_3 z^2 z'}{\bar{c}_z b_z} + \frac{x_G N_3}{b_K} \frac{z^2 z'}{\bar{c}_z} - x_{X_0} N_4 \bar{c}_z z'. \quad (4.26)$$

Equation of Euler [35], which in expanded/scanned form is written/recorded as follows, is condition of optimum $\bar{c}_z(z)$

$$\frac{\partial^2 F}{(\partial z')^2} z'' + \frac{\partial^2 F}{\partial z' \partial z} z' + \frac{\partial^2 F}{\partial z' \partial \bar{c}_z} - \frac{\partial F}{\partial z} = 0. \quad (4.27)$$

Having computed the derivatives provided by (4.27), we obtain

$$-2x_G N_3 \frac{z^2}{(\bar{c}_z)^2 b_z} - \frac{x_G N_3}{b_K} \frac{z^2}{(\bar{c}_z)^2} + x_{X_0} N_4 = 0. \quad (4.28)$$

Since functional (4.26) is linear relative to first-order derivative z', the solution (4.28) is degenerate and takes form of usual algebraic equation, where b_z - known function from z.

From (4.28) has unknown optimum law $\bar{c}_z(z)$

$$(\bar{c}_z)_{opt} = z\sqrt{\frac{\varkappa_G N_5}{\varkappa_{X_0} N_4}\left(\frac{2}{b_z} + \frac{1}{b_\kappa}\right)}. \qquad (4.29)$$

Is more convenient, however, to deal not concerning absolute values of coordinate z, but concerning relative $\bar{z} = \frac{z}{\frac{1-D_\Phi}{2}}$.

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Let us carry out appropriate replacement in formula (4.29). In addition to this, let us take b_κ and b_z in the form:

$$b_\kappa = \frac{2}{\tau+1}\frac{S}{l} = \frac{2}{\tau+1}\sqrt{\frac{G_0}{p_0\lambda}}; \quad b_z = \frac{2}{\tau+1}\sqrt{\frac{G_0}{p_0\lambda}}\,[1+(\tau-1)\bar{z}],$$

and in coefficient of N_4 let us take air density according to V. P. Vetchinkin's formula

$$\rho = \rho_0\frac{20-H}{20-H}.$$

where H in km.

Then form

$$(\bar{c}_z)_{opt} = \frac{\bar{z}}{2}\left(\sqrt{\frac{\lambda G_0}{p_0}} - D_\Phi\right)\sqrt{N_6\left[\frac{2}{1-(\tau-1)\bar{z}} + 1\right]}. \qquad (4.30)$$

where

$$N_6 = \frac{[illegible]\,10^{-3}\varphi p_0}{V_0}\,\frac{(20-H)}{(20-H)}\left(\frac{\varkappa_G}{\varkappa_{X_0}}\right)\sqrt{\frac{\lambda}{G_0}}\left(1-D_\Phi\sqrt{\frac{p_0}{\lambda G_0}}\right)[\lg Re]^{[illegible]}. \qquad (4.31)$$

will take solutions (4.29) after conversions.

In formulas (4.30) and (4.31) span $l = \sqrt{\frac{\lambda G_0}{p_0}}$;

G_0 - in kg, p_0 - in kgf/m², H - in km, V - in m/s, D_z - m.

Within limits of one class of aircraft (lg Re) [2.58] it is changed insignificantly and it is possible to take as its equal to: 180 - for aircraft of short distance; 200 - for aircraft of medium distance; 220 - for aircraft of long range.

With $\bar{z}$=1 we have $(\bar{c}_0)_{opt}$:

$$(\bar{c})_{\bar{z}=1}=[\bar{c}_0]_{opt}=\frac{1}{2}\left(\sqrt{\frac{\lambda G_0}{p_0}}-D_\Phi\right)\sqrt{N_c\left(\frac{2}{\eta}-1\right)}. \qquad (4.32)$$

From formulas (4.32) and (4.31) it is evident that with increase/growth $q, \varkappa_G, H, \lambda$ the optimum wing chord ratio at the root (along side the fuselage) increases; with increase $V, \varkappa_{\lambda_c}, D_\Phi$ in value $(\bar{c}_0)_{opt}$ they are reduced; with increase in p_0 of value $(\bar{c}_0)_{opt}$ they are reduced; with increase in G_0 of value $(\bar{c}_0)_{opt}$ they increase.

Thus, if increase in parameter leads to increase in weight of wing, then this increase leads also to increase $(\bar{c}_0)_{opt}$ and vice versa.

It must be noted that solution (4.30) does not depend on sweepback of wing, since area S when $(b_z, \bar{z})=\text{const}$ does not depend on χ, if chords are taken along flow.

The fact that with $\bar{z}$=0 (at wing tip) optimum relative thickness also must be equal to zero, is characteristic feature of obtained solution. This is explained by the fact that the bending moment at the wing tip is equal to zero. By the way, if we consider not only bending strength of wing, but also for the torsion, then the special feature of the solution indicated will not be changed, since the torsional moment at the wing tip is also equal to zero.

Virtually, however, it cannot be done $\bar{c}_{z=l}=0$, since necessary rigidity of end cross sections for torsion (it is feasible reversal of ailerons and wing flutter) will not be achieved/reached. In addition to this, when $\bar{c}_z \leqslant 0.06$ the lift effectiveness of profiles/airfoils sharply fall, the high-lift device of wing hinders, appear other structural/design difficulties. Taking into account the satisfaction of all these requirements is chosen value $\bar{c}_r$.

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Thus, formula (4.30) can be in practice used only in root wing sections, to 30-35% of span, and for remaining part of wing should be accepted either $\bar{c}_z=\bar{c}_r=0.08-0.09=\text{const}$. or law, close to linear [1], in this case at wing tip it is necessary to ensure $\bar{c}_r=0.08-0.09$.

FOOTNOTE [1]. Since it is necessary to consider the required volume of wing for positioning the fuel/propellant. ENDFOOTNOTE.

Examples of theoretical optimum law $\bar{c}_z(\bar{z})$, constructed according to formula (4.30), and practical optimum law, where in end and wing center sections $\bar{c}_z=\bar{c}_K=\text{const}$, they are given in Fig. 4.10.

Besides case examined above, when medium design stresses $\sigma_{cp}=\text{const}$ on wingspan, is also of interest case, characteristic for sweptback wings, when $\frac{(\sigma_{cp})_z}{c_z}=\text{const}=d$ on spread/scope. The solution in this case does not differ in principle from (4.29)-(4.30) and takes the form

$$(\bar{c}_z)_{cp}=\sqrt[3]{\frac{2z^2}{b_z}\left(\frac{2}{b_z}+\frac{1}{b_K}\right)\frac{E_8}{\varkappa_{X_0}N_4}}, \qquad (4.33)$$

where $E_8=\frac{4\eta\varphi G_0 n_f \varkappa_G}{3\xi f d l(\tau_i+1)}=\text{const};$

$\varkappa_{X_0}N_4=\text{const}.$

Comparison shows that under condition $(\sigma_{cp})_z/c_z=\text{const}$ optimum values $\bar{c}_z$ on the average on (5-7) % are more than condition $(\sigma_{cp})_z=\text{const}$.

Thus, variational problem examined gives possibility to justify theoretically and to obtain virtually (with correction for guarantee of rigidity of wing) optimum law of change in wing chord ratio on spread/scope.

Analogous variational problems can be solved and in connection with supersonic aircraft during determination of optimum laws $\bar{c}(z)$ of wing, sectional areas of fuselage along the length and the like taking into account change in aerodynamic, weight and rigid characteristics of aggregates/units with some assigned magnitudes, for example, with assigned volume for positioning fuel/propellant.

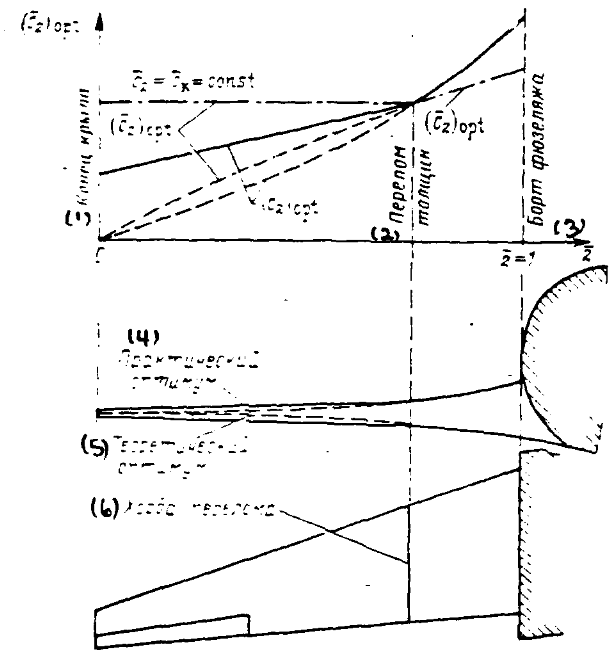

Fig. 4.10. Schematic of change in optimum values of relative and greatest absolute thickness of tapered wing on spread/scope: ---- - theoretical optimum taking into account bending strength of wing; —— - practical optimum taking into account rigidity of wing to torsion and bending.

Key: (1). Wing tip. (2). Fracture of thicknesses. (3). Side of fuselage. (4). Practical optimum. (5). Theoretical optimum. (6). Chord of fracture.

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Chapter V.

METHOD OF GRADIENTS OF TAKEOFF WEIGHT AND ITS APPLICATION DURING THE DESIGN OF AIRCRAFT.

Method of gradients of takeoff weight during design of aircraft relates to number of those approximated; however, in contrast to methods, examined in Chapter IV, it has specific special features and very great possibilities. Therefore it is expedient to consider this method in more detail.

§1. Description of method.

Gradient of takeoff weight is expression

$$\operatorname{grad} G_0 = \frac{\partial G_0}{\partial i}\Delta i = \varkappa_i \Delta i.$$

where Δi - increase in parameter or characteristic, for example, further weight of part or assembly, increase in aerodynamic characteristic, characteristic of engine and the like;

$\varkappa_i$ - derivative of takeoff weight from any parameter i.

Basic purpose of gradient of takeoff weight consists in the fact that with its aid it is possible to comparatively easily and rapidly count over original value of takeoff weight and to obtain new

$$(G_0)_{нов} = (G_0)_{исх} \pm \left(\frac{\partial G_0}{\partial i}\right) \Delta i. \qquad (5.1)$$

It is important to note that derivative $\varkappa_i$ is constant value for projected/designed aircraft in each stage of design.

Degree of error during determination of value $(G_0)_{нов}$ depends on absolute value of increase in parameter Δi (Fig. 5.1).

If value Δi does not exceed 10% of initial value of parameter, which frequently occurs at examination of different modifications of characteristics and parameters of aircraft, then with the help of linearization of equation $G_0(\Delta i)$ for formula (5.1) value $(G_0)_{нов}$ can be determined with an accuracy to 1%. This accuracy during the approximate computations usually completely satisfies designer.

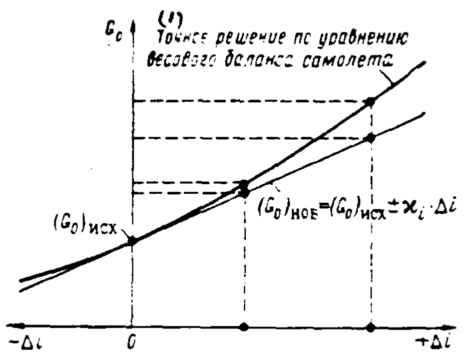

Fig. 5.1. To determination of takeoff weight with the help of gradient $\varkappa_i\cdot\Delta i$

Key: (1). Exact solution by the equation of the weight balance of aircraft.

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Method of gradients of takeoff weight it is possible to use not only for conversion of takeoff weight, but also for solutions of series/row of other problems. The designation/purpose of this method, generally speaking, consists in the fact that with its aid it is possible to connect initial (particular) changes in parameters and characteristics of aircraft components with final (total) changes in the evaluation criterion during some known limitations, for example, with constant/invariable basic flight-performance data, with the constant/invariable structural strength, etc.

Let us designate basic criterion of evaluation of aircraft through a and we will consider it as function of n quantity of variable parameters i. With infinitesimal change arbitrary (independent variables) the variables i total differential of evaluation criterion is equal to

$$da = \frac{\partial a}{\partial i_1} di_1 + \frac{\partial a}{\partial i_2} di_2 + \cdots + \frac{\partial a}{\partial i_n} di_n.$$

Assuming/setting with Δi→0 differentials and increases they are equivalent to mi, it is possible to register

$$\Delta a = \frac{\partial a}{\partial i_1} \cdot \Delta i_1 + \frac{\partial a}{\partial i_2} \cdot \Delta i_2 + \cdots + \frac{\partial a}{\partial i_n} \cdot \Delta i_n. \quad (5.2)$$

It is possible to use sum (5.2) in the case of the approximate calculations. Here Δa and Δi - finite (low) increments.

For example, during research of effect on prime cost t-km changes in weight of aircraft components ΔG, aerodynamic drag ΔX, specific hourly consumption of fuel/propellant Δc_p, resources/lifetimes ΔT and costs/values of aircraft components ΔC equation (5.2) takes form

$$\Delta a = \frac{\partial a}{\partial G} \cdot \Delta G + \frac{\partial a}{\partial X} \cdot \Delta X + \frac{\partial a}{\partial c_p} \cdot \Delta c_p + \frac{\partial a}{\partial T} \cdot \Delta T + \frac{\partial a}{\partial C} \cdot \Delta C. \quad (5.3)$$

Equation (5.3) gives possibility to solve following problems:

1) to determine effect on criterion of evaluation of aircraft of each of variable parameters, to detect/expose, which of them has greatest effect in order to direct efforts/forces first of all to improvement in this value. For the solution of this problem it is necessary to find the appropriate partial derivatives and to consider

that finite increments $(\Delta G, \Delta X, \Delta c_T)$ variables they must compose identical, comparatively small, the part of the initial value of each of them (for example, 1%);

2) to determine total, often contradictory, effect on the criterion of evaluation of any quantity of values. This problem has practical interest during the analysis of the diverse variants of the technical solutions, connected with a change in such values as G, X, T, etc. It is obvious that best will be that version of the solution, for which occurs the extremum of the sum of increases in the prime cost t-km, i.e.

$$[\Sigma - \Delta a]_{max} \quad \text{or} \quad [\Sigma + \Delta a]_{min}$$

3) to find the equivalents of different pairs of values, for example the weight equivalents of cost/value or lifetime of aggregate/unit, etc. This problem has very great practical value during the research of the profitability of the structural-design solutions, connected, for example, with the application of easier, but more expensive construction/design, with the decrease of aerodynamic drag due to a gain in weight, etc. For the solution of such problems it is necessary to equate to zero pair of any particular increases in the prime cost t-km (this problem is examined in Chapter XIV). Let us note that in the general case, using the formula

$$\sum_{i>2} \left(\frac{\partial a}{\partial x_k} \cdot \Delta x_k \right) = 0. \quad (5.4)$$

it is possible to find the equivalents of any quantity k of the variable parameters i, which interest designer.

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Thus, equation (5.3) allows with smaller labor inputs in comparison with routine calculation of new versions of aircraft to find in process of designing connection/communication between particular and total changes in characteristics, to compare between themselves particular changes in characteristics with identical final result, to give evaluation of profitability of constructive solutions.

Problems indicated can be solved in any stage of design of experimental model or during modifications of already constructed and operable aircraft. In this case the limitations (solution condition) of problems at each stage can be different. For example, at the stage of preliminary investigations and sketch design of experimental aircraft basic limitations appear as assigned parameters $(L;$ [illegible] $G_{..})$=const so also usually those taken $(p_0, \bar{P}_0)$=const. At the latter stage of design or during the modification of the existing sample of the aircraft, when characteristics of engines are determined, sizes/dimensions and the form of aircraft, as limitation can serve (S, P_0)=const. Furthermore, at this point it is known, what flight-performance data must be preserved with a change in other characteristics of aircraft. For example, upon the appearance of further weight of aggregate/unit or part, when it is necessary to preserve not only structural strength and calculated flying range, but also cruising speed, an increase in the takeoff weight and prime cost t-km will be different with (p_0, $\bar{P}_0$)=const and with (S, P_0)=const.

In this chapter application of method of gradients of takeoff weight assumes retention/maintaining cruising speed and calculated (practical) flying range.

These limitations during variation by weight and by aerodynamic characteristics of aircraft (taking into account $G_{п.н}=const$) imply very important corollary: economic criterion of evaluation of aircraft can be replaced with simpler criterion - takeoff weight. In this case the rational structural-design solution will correspond to the minimum of takeoff weight.

One should note that takeoff weight is not too narrow a criterion, since its value according to equation of weight balance is organically-bound with weight of aircraft components, aerodynamic engine characteristics. If are retained $V_{крейс}$, $L_{расч}$ and $G_{п.н}$, then takeoff weight can serve as the criterion of evaluation of the aircraft of different designation/purpose.

Important value has the fact that during use of weight criterion solution of designing problems becomes considerable compact and it is simpler than during use of criteria, based on efficiency/cost-effectiveness or combat efficiency of aircraft.

Knowing derivatives of takeoff weight $\varkappa_i$, it is possible to find total change in takeoff weight ΔG_0 as evaluation criterion with change in characteristics of aircraft components:

$$\Delta G_0=\frac{\partial G_0}{\partial G^*}\cdot\Delta G^*+\frac{\partial G_0}{\partial X_0}\cdot\Delta X_0+\frac{\partial G_0}{\partial c_p}\cdot\Delta c_p+\ldots=$$
$$=\varkappa_G\cdot\Delta G^*+\varkappa_{X_0}\cdot\Delta X_0+\varkappa_{c_p}\cdot\Delta c_p+\ldots \qquad (5.5)$$

Here $\varkappa_G=\partial G_0/\partial G^*$ - the derivative of takeoff weight by the further weight of any aircraft component (G* - the further weight of part, assembly or aggregate/unit);

$\varkappa_{X_0}=\partial G_0/\partial X_0$ - derivative of takeoff weight on aerodynamic drag of aircraft, etc average/mean during the flight.

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Let us pause at determination of some most important derivatives of takeoff weight in simplest cases, when only any particular characteristic of aggregate/unit (for example, weight) is changed, and remaining particular characteristics (aerodynamic, volume, etc.)

remain constant/invariable.

S2. Determination of the derivatives of takeoff weight from the additional weight of aircraft components.

In process of designing aircraft following task frequently meets:

- it is decided to change weight of any aggregate/unit, group of equipment or load, after preserving aerodynamic and other characteristics of aircraft. How in this case it is necessary to change takeoff weight so that the fundamental flight characteristics and the strength of aircraft would remain as before?

Such problems are characteristic for stage of sketch design, when on basis of variations in different parameters basic dimensions are chosen and weight of aircraft is calculated. However, at the subsequent stages of the development of design (even during the modifications of the constructed aircraft) these problems are not excluded, since the detailed study of the construction of the aircraft is usually accompanied by some deviations of the weight of aggregates/units from their values, established/installed during the sketch design. For example, if the construction of the aircraft is obtained heavier than it was assumed, then as the compensation it is possible to decrease either the fuel load (reducing the flying range), or payload weight, after keeping takeoff weight constant, or to increase takeoff weight, after preserving flying range, cruising speed and payload, but in the latter case after worsening/impairing takeoff and landing characteristics, if there is no possibility to raise the engine thrust.

Let us assume that at any stage of design of aircraft need for introducing further weight G^{*} appeared. If with a change in the weight of any unit on $G_1{*}$ an even further weight $G_2{*}$ for the correction of a possible change in the position of center of gravity in aircraft, increase in the rigidity of construction/design, etc. is necessary, then we will consider that the further weight G* is the sum of all these components, i.e., $G{*}=G_1{*}+G_2{*}+ \ldots$.

Let be known dependences of weights of all components on takeoff weight of aircraft, and also value, which designer considers necessary to leave constant/invariable, for example, $(p_0, \bar{P}_0)$=const, or (S, P_0)=const, or $\bar{G}_T$=const and so forth, etc.

It is necessary to determine increase in takeoff weight, necessary for accomplishing limitations (on which depends retention/maintaining flight characteristics) presented, and also for retaining/maintaining structural strength.

Let us register equation of weight balance of aircraft upon appearance of further weight G*

$$G_0 = G_{пуст} + G_т + G_{п.н} + G_{служ} + G^*, \quad (5.6)$$

where $G_{пуст}$ - empty weight, $G_{пуст} = f_1(G_0)$;

$G_{п.н}$ - payload weight, $G_{п.н} = \mathrm{const}$;

$G_{служ}$ - weight of official load, $G_{служ} = \mathrm{const}$;

$G_т$ - fuel load, $G_т = f_2(G_0)$;

G^* - further weight.

From (5.6) it follows that

$$G^* = G_0 - G_{пуст} - G_т - G_{п.н} - G_{служ}.$$

Let us take partial derivative on G_0

$$\frac{\partial G^*}{\partial G_0} = 1 - \frac{\partial}{\partial G_0}(G_{пуст} + G_т).$$

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Since it is necessary to determine derivative $\partial G_0/\partial G^*$, which shows effect of G* on G_0, then on the basis of previous we have

$$\varkappa_G = \frac{\partial G_0}{\partial G^*} = \frac{1}{1 - \frac{\partial}{\partial G_0}(G_{пуст} - G_т)}. \quad (5.7)$$

New value of takeoff weight of aircraft is determined with the help of $\varkappa$ according to formula [1]

$$(G_0)_{нов} = (G_0)_{исх} \pm \varkappa_G G^*. \quad (5.8)$$

FOOTNOTE [1]. In the literature derivative $\varkappa_G$ is known by the name "coefficient of increase/growth". ENDFOOTNOTE.

Here $(G_0)_{исх}$ - initial takeoff weight with G*=0; plus sign before second term is taken in the case of an initial gain in weight (+G*), and minus - in the case of decreasing weight (-G*).

Numerical values $\varkappa_G$ can be changed, generally speaking, from one to. The latter is possible, when $\frac{\partial}{\partial G_0}(G_{пуст}+G_т)=1$ [see formula (5.7)]. But if

$$\frac{\partial}{\partial G_0}(G_{пуст}-G_т)=0, \text{ then } \varkappa_G=1. \text{ In this case}$$

$$(G_{пуст}, G_т)=\mathrm{const} \text{ and } (G_0)_{нов}=(G_0)_{исх}\pm G^*,$$

which corresponds to the case, when all aircraft performance and strength of its aggregates/units [2] are changed.

FOOTNOTE [2]. This case occurs during the modification of the constructed aircraft. ENDFOOTNOTE.

In general case one part of weight components of aircraft depends on takeoff weight, moreover all functions $G_i(G_0)$ are known and limitations, placed on them, and another part of weight components remains constant. Then the equation of weight balance (5.6) it is possible to write thus:

$$G_0=\sum_{1}^{m} G_i(G_0)-\sum_{m+1}^{n} G_k+G^*,$$

where $\sum_{m+1}^{n} G_k = \text{const}$.

If we perform the same operations, that also during derivation of formula (5.7), we will obtain in general case

$$\varkappa_G=\frac{\partial G_0}{\partial G^*}=\frac{1}{1-\frac{\partial}{\partial G_0}\left[\sum_{1}^{m} G_i(G_0)\right]}. \tag{5.9}$$

Usually functions $G_i(G_0)$ are weight formulas, for example, wing, fuselage, to chassis/landing gear, power plant, fuel/propellant, etc. Since all derivatives $\partial G_i/\partial G_0 \geqslant 0$, obviously, the greater the value and a number of these derivatives, the greater and the value $\varkappa_G$. Let us note that values $\partial G_i/\partial G_0$ depend not only on the form of the function $G_i(G_0)$, but also on the character of the limitations, placed on parameters of aircraft. Among all limitations, as it was mentioned above, of greatest interest are the limitations of form (p_0, $\bar{P}_0$)=const or (S, P_0)=const, with which are retained all or the part of aircraft performance.

It is important to also note that value $\varkappa_G$ can be more than than one only in such a case, when with introduction of further weight G* designer attempts to keep constants any assigned properties of aircraft: flight-performance data, strength, etc.

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Let us consider physical essence of emergence of derivative $\varkappa_G$, when $\varkappa_G > 1$.

Let us assume that on projected/designed aircraft it was necessary to establish/install auxiliary equipment by weight G*, which does not worsen/impair aerodynamics of aircraft and does not require supplementary volumes. It is necessary, after preserving fundamental flight characteristics, payload, and also the strength of aggregates/units, to modify aircraft. For retaining/maintaining flight characteristics, obviously, it is necessary to preserve load on 1 m^2 of wing p_0 and thrust-weight ratio $\bar{P}_0$ during the takeoff.

With gain in weight of aircraft by G* will increase and load on wing, fuselage, chassis/landing gear and tail assembly. Consequently, for retaining/maintaining the strength of these aggregates/units it is necessary to increase sizes/dimensions and weight of their load-bearing elements. Further, for satisfaction of condition $p_0=G_0/S=const$ it is necessary with change G_0 to respectively increase the sizes/dimensions of wing, and when $\bar{P}_0=const$ - and the thrust of engines (if it is not possible to increase thrust by other methods), which by itself will involve a gain in weight of wing and power plant. Further, with increase of the sizes/dimensions of wing, it is necessary to increase also sizes/dimensions and weight of tail assembly for keeping constant/invariable the stability characteristics of aircraft. Finally, for retaining/maintaining the flying range it will be required, obviously, to increase the initial fuel reserve, taking into account condition $\bar{G}_T = G_T/G_0 = \text{const}$. The given gain in weight in the separate aircraft components and initial fuel reserve will lead to the increase in the takeoff weight of aircraft, which, in turn, will involve an increase in the load on the aggregates/units and an increase in their weight. As a result will be required new, secondary, increase of the weight of aggregates/units and fuel/propellant.

Process is finished then, when takeoff weight reaches value, obtained from solution of equations of weight balance (5.6) or (5.8).

Thus, increase in weight of aircraft G* raises takeoff weight by considerably larger value $\varkappa_G G^*$.

Derivative $\partial G_{\text{пуст}}/\partial G_0$ entering formula (5.7), is determined from formula

$$\frac{\partial G_{\text{пуст}}}{\partial G_0} = \frac{\partial G_{\text{кр}}}{\partial G_0} + \frac{\partial G_{\text{ф}}}{\partial G_0} + \frac{\partial G_{\text{оп}}}{\partial G_0} + \frac{\partial G_{\text{ш}}}{\partial G_0} + \frac{\partial G_{\text{с.у}}}{\partial G_0} + \frac{\partial G_{\text{об.упр}}}{\partial G_0}.$$

Here $G_{\text{кр}}, G_{\text{ф}}, G_{\text{оп}}, G_{\text{ш}}, G_{\text{с.у}}, G_{\text{об.упр}}$ - weight of wing, fuselage, tail assembly, chassis/landing gear, power plant, equipment and control respectively.

Weight formulas of wing and other aggregates/units depending on G_0 are usually known.

During determination of derivative $\partial G_T/\partial G_0$ also entering in (5.7), one should consider that fuel load is linearly connected with takeoff weight, i.e.

$$\frac{\partial G_т}{\partial G_0} = \overline{G}_т = \frac{G_т}{G_0}. \quad (5.10)$$

For approximate computations it is possible to utilize following dependence, which escapes/ensues from known formula of Breguet, which determines flying range,

$$\frac{\partial G_т}{\partial G_0} = \overline{G}_т = 1 - e^{-\frac{L_{расч}(c_p)_{ср}}{V_{рейс}K_{ср}}} + \overline{G}_{к.з}. \quad (5.11)$$

If $\overline{G}_т \leqslant 0.2$, then formula (5.11) can be simplified.

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Expanding second term in the series/row and being limited two first terms of expansion, we will obtain with error ±2%

$$\frac{\partial G_т}{\partial G_0} = \overline{G}_т \approx \frac{1.1 L_{расч}(c_p)_{ср}}{V_{рейс}K_{ср}} + \overline{G}_{к.з}, \quad (5.12)$$

where 1.1 - correction factor.

Derivative $\varkappa_G$ approximately can be determined from principle of linear similarity [20] according to formula

$$\varkappa_G = \frac{1}{1 - \overline{G}_т - \frac{k_п}{2}\overline{G}_{п.уст}}. \quad (5.13)$$

where $k_п$ - coefficient of linear similarity ($k_п = 2.2-2.6$).

Table 5.1 gives computed values of derivative of takeoff weight $\varkappa$ for passenger aircraft with TVRD.

From data of Table 5.1 it is evident that with (S, P_0)=const value $\varkappa_G$ is considerably less than when $(p_0, \bar{P}_0)=\text{const}$. This is explained by the fact that the supplementary weight G* when $(p_0, \bar{P}_0)=\text{const}$ implies a gain in weight of all entering G_0 components, except commercial $G_{\text{ком}}$ and official $G_{\text{служ}}$ of loads, since usually $(G_{\text{ком}}, G_{\text{служ}})=\text{const}$. With (S, P_0)=const the weight of power plant and the part of the weight of wing, which depends on its area, remain constant/invariable.

If supplementary weight G* does not imply need for correction of position of center of gravity in aircraft, it does not require increase in volume of any of aggregates/units and its appearance does not change directly aerodynamic characteristics of aircraft, then numerical value $\varkappa_G$ does not depend on origin of supplementary weight G*. In whatever aircraft component or its equipment was formed supplementary weight, a change in the takeoff weight will be identical and equal to $\Delta G_0 = \pm\varkappa_G G^*$.

However, it is far significant, was formed supplementary weight G* due to payload, fuel/propellant or empty aircraft. In all these cases, with equal values $\varkappa_G$ the value of the general/common criterion of evaluation of aircraft (for example, efficiency/cost-effectiveness or of combat efficiency) will be different. In this sense the derivative of takeoff weight $\varkappa_G$ is very multifaceted.

Role of derivative $\varkappa_G$ not only in the fact that it simplifies determination of new takeoff weight at any stage of design, but also it makes it possible to judge weight perfection of projected/designed aircraft. It is obvious that the greater the value $\varkappa_G$. the more attentive must be the relation to the weight perfection of aircraft, the more is required the expenditures of time and resources for the reduction of the weight of each aggregate/unit, assembly and part.

Table 5.1. Values of the derivative of takeoff weight $\varkappa_G$ of passenger aircraft.

(1) Тип самолета	(2) Значения $\varkappa_G$		(4) Практическая дальность полета при максимальной коммерческой нагрузке в км
	(3) при условии $(p_0, \bar{P}_0) =$ = const	(3) при условии $(S, \bar{P}_0) =$ = const	
Местных авиалиний (5)	2,1—2,2	1,5—1,6	600—1200
Средний магистральный (6)	2,3—2,6	1,7—1,8	1600—2200
Тяжелый магистральный (7)	3,2—3,8	2,2—2,6	4500—5500
Сверхзвуковой (8)	8—10	3,5—4,0	6000—6500

Key: (1). Aircraft type. (2). Values. (3). under condition. (4). Practical flying range with maximum payload in km. (5). Feeder lines. (6). Average/mean main-line. (7). Heavy main-line. (8). Supersonic.

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§ 3. Determination of the derivatives of takeoff weight from the aerodynamic characteristics of aircraft.

Let us consider how lift-drag ratio **K** affects, drag $\mathbf{X_0}$ (when $c_y=0$) and coefficient c_{x0} by takeoff weight of aircraft.

Basic initial conditions as before: $(L_{расч}, G_{п.н})=\text{const}$, and also $(p_0, \bar{P}_0)=\text{const}$ or $(S, \bar{P}_0)=\text{const}$. Is not difficult to show that with variations X_0, K and c_{x0} cruising speed due to change in altitude of flight [1] can remain constant, i.e., $V_{крейс}=\text{const}$, what indicates $V_{рейс}=\text{const}$, since $L_{расч}=\text{const}$.

FOOTNOTE [1]. A small change of all values, including flight altitude is assumed. In this case $V_{кр}=V_{рейс}\approx\text{const}$. ENDFOOTNOTE.

Problem consists of similar to that found in § 2 derivatives ∂G₀/∂G* to determine $\partial G_0/\partial K,\ \partial G_0/\partial X_0,\ \partial G_0/\partial c_{x0}$.

For this we will use equation of weight balance of aircraft in relative values (all weight components are related to value of takeoff weight)

$$1=\overline{G}_{пуст}+\overline{G}_{т}+\overline{G}_{п.н}+\overline{G}_{служ},$$

where $\overline{G}_{т}=\overline{G}_{т.расх}+\overline{G}_{н.з}=1-e^{-\frac{L_{расч}(c_e)_{ср}}{V_{рейс}K_{ср}}}+\overline{G}_{н.з}$;

$K_{ср}$ and $(c_e)_{ср}$ - average/mean for flight time of value of lift-drag ratio and specific hourly consumption of fuel/propellant.

Consequently, we have

$$1=\overline{G}_{пуст}+\overline{G}_{п.н}+\overline{G}_{служ}+\overline{G}_{н.з}+1-e^{-\frac{L_{расч}(c_e)_{ср}}{V_{рейс}K_{ср}}}.$$

From this equation average/mean required value of lift-drag ratio is equal

$$K_{ср}=-\frac{L_{расч}(c_e)_{ср}}{V_{рейс}\ln(\overline{G}_{пуст}+\overline{G}_{п.н}+\overline{G}_{служ}+\overline{G}_{н.з})}. \tag{5.14}$$

Lift-drag ratio is here not only explicit function of values, which stand in right side of equality (5.14), but it depends also on sizes/dimensions of aircraft [2].

FOOTNOTE [2]. The manifestation of "scale effect", if change G_0 is connected with a change in the sizes/dimensions of aircraft. ENDFOOTNOTE.

Lift and descending of subsonic aircraft are conducted, as a rule, with lift-drag ratio, which differs little from cruise. Therefore for the subsonic aircraft with a sufficient accuracy it is possible to accept $K_{\text{ср}} \approx K_{\text{крейс}}$.

Let us take partial derivative $K_{\text{ср}}$ on G_0 and will write it in the form of reverse/inverse, interesting us, value $\partial G_0/\partial K_{\text{ср}}$. As a result we will obtain, counting $\bar{G}_{\text{н.з}}=\text{const}$.

$$\varkappa_K = \frac{\partial G_0}{\partial K_{\text{ср}}} =$$

$$= \left\{ \frac{L_{\text{расч}}(c_p)_{\text{ср}}}{V_{\text{рейс}}} \frac{\frac{\partial}{\partial G_0}(\bar{G}_{\text{пуст}} + \bar{G}_{\text{п.н}} + \bar{G}_{\text{служ}})}{[\ln(\bar{G}_{\text{пуст}} + \bar{G}_{\text{п.н}} + \bar{G}_{\text{служ}} + \bar{G}_{\text{н.з}})]^2 (\bar{G}_{\text{пуст}} + \bar{G}_{\text{п.н}} + \bar{G}_{\text{служ}} + \bar{G}_{\text{н.з}})} - \right.$$

$$\left. - \left(\frac{\partial K_{\text{ср}}}{\partial G_0} \right)_{\bullet} \right\}^{-1}. \tag{5.15}$$

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Here $(\partial K_{\text{ср}}/\partial G_0)_{\bullet}$ is expressed "scale effect", when together with change in weight of aircraft G_0 change its sizes/dimensions [under condition $(p_0, \bar{P}_0) = \text{const}$] [1].

FOOTNOTE [1]. For the subsonic aircraft in the approximate computations it is possible to accept $K_{\text{ср}} = 12.5 - 0.0331\, G_0^{0.344}$ and, consequently, $(\partial K_{\text{ср}}/\partial G_0)_{\bullet} = 0{,}0114\, G_0^{-0.656}$, where G_0 is taken in kg; for supersonic

$(\partial K_{cp}/\partial G_0)_* \approx 0.173 \cdot G_0^{-1}$, where G_0 in t. ENDFOOTNOTE.

In the case, when sizes/dimensions of aircraft do not change with change in lift-drag ratio, i.e., under condition $(S, P_0)=\text{const}$, derivative $(\partial K_{cp}/\partial G_0)_*=0$ and

$$\varkappa_K = \frac{V_{\text{рейс}}}{(c_p)_{cp} L_{\text{расч}}} \frac{[\ln(\bar{G}_{\text{пуст}} + \bar{G}_{\text{п.н}} + \bar{G}_{\text{служ}} + \bar{G}_{\text{н.з}})]^2 (\bar{G}_{\text{пуст}} + \bar{G}_{\text{п.н}} + \bar{G}_{\text{служ}} + \bar{G}_{\text{н.з}})}{\frac{\partial}{\partial G_0}(\bar{G}_{\text{пуст}} + \bar{G}_{\text{п.н}} + \bar{G}_{\text{служ}})}. \quad (5.16)$$

If over-all payload ratio of expendable fuel/propellant $\bar{G}_{\text{т расх}} \leqslant 0.2$, then with accuracy ±2%

$$1 - e^{-\frac{L_{\text{расч}}(c_p)_{cp}}{V_{\text{рейс}} K_{cp}}} \approx \frac{1,1 L_{\text{расч}}(c_p)_{cp}}{V_{\text{рейс}} K_{cp}}$$

and formula for $\varkappa_K$ is reduced to simpler form

$$\varkappa_K = \frac{V_{\text{рейс}}}{1,1 L_{\text{расч}}(c_p)_{cp}} \frac{(1 - \bar{G}_{\text{пуст}} - \bar{G}_{\text{п.н}} - \bar{G}_{\text{служ}} - \bar{G}_{\text{н.з}})^2}{\frac{\partial}{\partial G_0}(\bar{G}_{\text{пуст}} + \bar{G}_{\text{п.н}} + \bar{G}_{\text{служ}})}. \quad (5.17)$$

Let us find now how changes takeoff weight with change in aerodynamic drag of aircraft X_0 (when $c_y=0$) [2].

FOOTNOTE [2]. Here is intended force $X_0 = c_{x0} S q \approx (X_0)_{cp}$ average/mean for the flight time since $(c_{x0}, q) \approx \text{const}$ during entire flight. ENDFOOTNOTE.

In this case, as it is not difficult to show, it is possible to preserve flight speed, if $(c_x, c_y, \rho, \lambda) = \text{const}$ with M<1, or $(c_x, c_y, \rho, c_y^{\alpha}) = \text{const}$ with M>1.

Partial derivative $\partial G_0/\partial X_0$ is expressed as $\varkappa_K$:

$$\frac{\partial G_0}{\partial X_0}=\frac{\partial G_0}{\partial K_{cp}}\cdot\frac{1}{(\partial X_0\,\partial K_{cp})}=\frac{\varkappa_K}{\partial X_0\,\partial K_{cp}}. \quad (5.18)$$

At subsonic speed

$$X_0=X-X_i=c_xSq-c_{xi}Sq=\frac{G_{cp}}{K_{cp}}-\frac{G_{cp}K_{cp}c_x}{\pi\lambda_{\text{эф}}},$$

where G_{cp} - average for the flight time weight of aircraft;

$$G_{cp}\approx G_0-\frac{1}{2}G_{\text{т расх}};$$

$$(\lambda_{\text{эф}},\ c_x)=\text{const.}$$

With these conditions and number M<1

$$\frac{\partial X_0}{\partial K_{cp}}=-G_{cp}\left(\frac{1}{K_{cp}^2}+\frac{c_x}{\pi\lambda_{\text{эф}}}\right), \quad (5.19)$$

while with number M>1

$$\frac{\partial X_0}{\partial K_{cp}}=-G_{cp}\left(\frac{1}{K_{cp}^2}+\frac{c_x}{c_y^a}\right). \quad (5.20)$$

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Substituting in (5.18) $\partial X_0/\partial K_{cp}$ from (5.19) or (5.20), we will obtain with number M<1

$$\frac{\partial G_0}{\partial X_0}=\varkappa_{X_0}=-\frac{\varkappa_K}{G_{cp}\left(\frac{1}{K_{cp}^2}-\frac{c_x}{\pi\lambda_{\text{эф}}}\right)}; \quad (5.21)$$

with number M>1

$$\frac{\partial G_0}{\partial X_0}=\varkappa_{X_0}=-\frac{\varkappa_K}{G_{cp}\left(\frac{1}{K_{cp}^2}-c_x\,c_y^a\right)}. \quad (5.22)$$

Is sometimes more convenient to determine change of takeoff weight not in dependence on change in aerodynamic drag X_0, but in dependence on change in its coefficient c_{x0}.

In this case, accepting, as earlier, with number M<1 $(c_x, c_y\rho, \lambda) = \text{const}$, and with number M>1 $(c_x, c_y\rho, c_y^{\alpha}) = \text{const}$, it is not difficult to obtain

$$\frac{\partial G_0}{\partial c_{x_0}} = \frac{\partial G_0}{\partial K_{\text{ср}}} \cdot \frac{\partial K_{\text{ср}}}{\partial c_{x_0}} = \varkappa_K \cdot \frac{\partial K_{\text{ср}}}{\partial c_{x_0}}. \tag{5.23}$$

With M<1 from the equation of polar we have

$$c_x = c_{x_0} + \frac{c_y^2}{\pi\lambda_{\text{эф}}},$$

whence $c_y = \sqrt{\pi\lambda_{\text{эф}}(c_x - c_{x_0})}$.

Consequently, with M<1

$$K_{\text{ср}} = \frac{c_y}{c_x} = \sqrt{\frac{\pi\lambda_{\text{эф}}}{c_x}\left(1 - \frac{c_{x_0}}{c_x}\right)}. \tag{5.24}$$

by analogy with M>1

$$K_{\text{ср}} = \sqrt{\frac{c_y^{\alpha}}{c_x}\left(1 - \frac{c_{x_0}}{c_x}\right)} \tag{5.25}$$

Taking into account that with number M<1

$$K_{\max} = \frac{1}{2}\sqrt{\frac{\pi\lambda_{\text{эф}}}{c_{x_0}}}$$

(here $K_{\max}$ - the original value of quality to change c_{x0}, with which $c_x - c_{x0} = c_{xi} = c_{x0}$), from (5.24) we have

$$\frac{\partial K_{\text{ср}}}{\partial c_{x_0}} = -\frac{1}{2c_x}\sqrt{\frac{\pi\lambda_{\text{эф}}}{c_x - c_{x_0}}} = -\frac{K_{\max}}{c_x}. \tag{5.26}$$

Respectively and with M>1 we obtain

$$\frac{\partial K_{cp}}{\partial c_{x_0}}=-\frac{1}{2c_x}\sqrt{\frac{c_p^a}{c_x-c_{x_0}}}=-\frac{K_{max}}{c_x}. \quad (5.27)$$

After substituting in (5.23) $\partial K_{cp}/\partial c_{x0}$ from (5.26) or (5.27) and after designating $\partial G_0/\partial c_{x_0}=\varkappa_{c_{x_0}}$, we will obtain with number M<1

$$\varkappa_{c_{x_0}}=-\frac{\varkappa_K}{2c_x}\sqrt{\frac{\pi\lambda_{ef}}{c_x-c_{x_0}}}=-\frac{\varkappa_K K_{max}}{c_x}. \quad (5.28)$$

with number M>1

$$\varkappa_{c_{x_0}}=-\frac{\varkappa_K}{2c_x}\sqrt{\frac{c_p^a}{c_x-c_{x_0}}}=-\frac{\varkappa_K K_{max}}{c_x}. \quad (5.29)$$

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Table 5.2 gives exemplary/approximate computed values of derivatives $\varkappa_{c_x}$, $\varkappa_x$ and $\varkappa_{c_{x_0}}$ for passenger aircraft:

From comparison of values $\varkappa_G$ and $\varkappa_X$ (Table 5.1 and 5.2) it is evident that increase X_0 on 1 kg implies, other conditions being equal, considerably larger increase in takeoff weight G_0, than overstress on 1 kg of any aggregate/unit. This conclusion/output is especially graphic for the heavy aircraft.

It must be noted that is feasible case, when change in weight of aggregate/unit or unit connected with change of its volume, where is known dependence of force of parasitic aerodynamic drag X_0 on volume v. Then formula for determination $\varkappa_G$ taking into account a change in the volume of aggregate/unit will take the form

$$\varkappa_G=\frac{\partial G_0}{\partial G^*}=\frac{\partial G_0}{\partial v}\cdot\frac{\partial v}{\partial G^*}=\frac{\partial G_0}{\partial X_0}\cdot\frac{\partial X_0}{\partial v}\cdot\frac{\partial v}{\partial G^*}\approx\varkappa_{X_0}\cdot\frac{\Delta X_0}{\Delta v}\cdot\frac{\Delta v}{\Delta G^*}. \quad (5.30)$$

Analogously can be obtained formulas for $\varkappa_G$ in cases, when change in weight and volume of aggregate/unit is connected with change in any other aerodynamic characteristics (lift-drag ratio, etc.).

§ 4. Determination of the derivatives of takeoff weight from the specific hourly consumption of fuel.

Of all engine characteristics of greatest interest during design of aircraft are: weight per horsepower (weight of engine, in reference to takeoff thrust), specific hourly consumption of fuel/propellant, service life and cost/value. Assuming that the weight of engine does not depend on the specific hourly consumption of fuel/propellant, it is possible to find $\partial G_0/\partial G_{\text{дв}}$ from formula (5.7).

Table 5.2. Values of the derivatives of takeoff weight $\varkappa_K$, $\varkappa_{\lambda_0}$, $\varkappa_{c_{x_0}}$.

(1) Назначение самолета	(2) Производные взлетного веса	(3) Условие $(p_0, \bar{F}) = \text{const}$	$(S, F) = \text{const}$
(4) Для местных авиалиний	$\varkappa_K$ в кгс/ед кач (5)	−(300—700)	−(200—400)
	$\varkappa_{\lambda_0}$ в кгс/кгс (6)	2,0—3,0	1,3—1,8
	$\varkappa_{c_{x_0}} \cdot 10^{-5}$	2—6	1,2—4,0
(7) Для средней дальности	$\varkappa_K$	−(2500—3000)	−(1700—1900)
	$\varkappa_{\lambda_0}$	11—12	7—8
	$\varkappa_{c_{x_0}} \cdot 10^{-5}$	25—30	17—19
(8) Для большой дальности	$\varkappa_K$	−(14·10³—19·10³)	−(7·10³—9·10³)
	$\varkappa_{\lambda_0}$	19—26	11—13
	$\varkappa_{c_{x_0}} \cdot 10^{-5}$	140—220	80—[illegible]
(9) Сверхзвуковые пассажирские самолеты	$\varkappa_K$	−(6·10⁴—7·10⁴)	−(3·10⁴—3,5·10⁴)
	$\varkappa_{\lambda_0}$	22—27	10—11
	$\varkappa_{c_{x_0}} \cdot 10^{-5}$	250—280	120—130

Key: (1). Designation/purpose of aircraft. (2). Derivatives of takeoff weight. (3). Condition. (4). For feeder lines. (5). in kgf/unit of performance. (6). in kgf/kgf. (7). For medium distance. (8). For long range. (9). Supersonic passenger aircraft.

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Lifetime and cost of engine affect the economy, but they do not affect takeoff weight of aircraft; therefore let us consider the effect only of the specific hourly consumption of fuel/propellant per the takeoff weight of aircraft.

From formula (5.14) it follows that

$$(c_p)_{cp} = -\frac{V_{рейс} K_{cp}}{L_{расч}} \ln (\overline{G}_{пуст} + \overline{G}_{п.н} + \overline{G}_{сл.уж} + \overline{G}_{н.з}) \qquad (5.31)$$

where $(c_p)_{cp}$ and K_{cp} - average/mean for flight time required values of specific hourly consumption of fuel/propellant and lift-drag ratio.

We accept, that

$$(L_{расч},\ G_{п.н},\ V_{рейс},\ \bar{G}_{н.з})=\text{const}.$$

If with variation $(c_p)_{cp}$, besides change in takeoff weight, must change also sizes/dimensions of aircraft, for example with imposition of condition $(p_c,\ \bar{P}_0)=\text{const}$, then $K_{cp}=K_{cp}(G_c)$ as a result of manifestation of "scale effect".

We differentiate (5.31) on G_0

$$\frac{\partial (c_p)_{cp}}{\partial G_0}=-\left[\frac{V_{рейс}K_{cp}}{L_{расч}}\ \frac{\frac{\partial}{\partial G_0}(\bar{G}_{пуст}+\bar{G}_{п.н}+\bar{G}_{служ})}{(\bar{G}_{пуст}+\bar{G}_{п.н}+\bar{G}_{служ}+\bar{G}_{н.з})}+\right.$$
$$\left.+\frac{V_{рейс}}{L_{расч}}\left(\frac{\partial K_{cp}}{\partial G_0}\right)_{*}\ln(\bar{G}_{пуст}+\bar{G}_{п.н}+\bar{G}_{служ}+\bar{G}_{н.з})\right].$$

Since we are not interested in $\frac{\partial (c_p)_{cp}}{\partial G_0}$, but inverse value, then from the previous we have:

$$\varkappa_{c_p}=\frac{\partial G_0}{\partial (c_p)_{cp}}=-\left[\frac{V_{рейс}K_{cp}}{L_{расч}}\ \frac{\frac{\partial}{\partial G_0}(\bar{G}_{пуст}+\bar{G}_{п.н}+\bar{G}_{служ})}{(\bar{G}_{пуст}+\bar{G}_{п.н}+\bar{G}_{служ}+\bar{G}_{н.з})}+\right.$$
$$\left.+\frac{V_{рейс}}{L_{расч}}\left(\frac{\partial K_{cp}}{\partial G_0}\right)_{*}\ln(\bar{G}_{пуст}+\bar{G}_{п.н}+\bar{G}_{служ}+\bar{G}_{н.з})\right]^{-1}. \qquad (5.32)$$

Approximate determination of derivative $\left(\frac{\partial K_{cp}}{\partial G_0}\right)_{*}$ (account of "scale effect") it is given above, into § 3.

If with change c_p sizes/dimensions of aircraft remain constants, for example when $(S,\ P_0)=\text{const}$, then $\left(\frac{\partial K_{cp}}{\partial G_0}\right)_{*}=0$ and formula (5.32) is simplified:

$$\varkappa_{c_p}=\frac{\partial G_0}{\partial (c_p)_{cp}}=-\frac{L_{расч}}{V_{рейс}K_{cp}}\left[\frac{\bar{G}_{пуст}+\bar{G}_{п.н}+\bar{G}_{служ}+\bar{G}_{н.з}}{\frac{\partial}{\partial G_0}(\bar{G}_{пуст}+\bar{G}_{п.н}+\bar{G}_{служ})}\right]. \qquad (5.33)$$

Typical values $\varkappa_{c_p}$ for commercial airplanes are given in Table .., where $\varkappa_{c_p}$ has dimensionality $\left[\frac{кгс}{ед\cdot c_p}10^3\right]$.

From Table 5.3 it is evident that when $(p_0; P_0)=\text{const}$ value $\varkappa_{c_f}$ more than with (S, P_0)=const. The reason for this is the same as the reason for difference $\varkappa_i$ with indicated limitations (see § 2).

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In conclusion let us do several resultant observations.

1. Having different derivatives of takeoff weight $\varkappa_i$, it is possible to determine new value of takeoff weight according to formula

$$G_{0\,new} = G_{0\,init} + \Sigma(\varkappa_i \Delta i),$$

where Δi - increase in any of characteristics.

2. Independence is characteristic to all derivatives of takeoff weight of sign of change of corresponding characteristics (invariance to sign Δi).

3. Besides antiderivatives of takeoff weight examined, can be analogously obtained all other interesting designer derivatives in each specific case.

4. Derivatives of takeoff weight are actually expression of feedback, since they give possibility to carry out interconnection of particular changes in characteristics of aircraft components with change in takeoff weight, which is criterion of evaluation of aircraft with retention/maintaining of flight-performance data (L_{range}, V_{cruise}) and payload.

Table 5.3. Values of the derivatives of takeoff weight $\varkappa_{c_p}$.

(1) Назначение самолета	(8) Условие	
	$(p_0, \bar{P}_0) = \mathrm{const}$	$(S, P_0) = \mathrm{const}$
(2) Для местных авиалиний	$\varkappa_{c_p} = 2-8 \left[\frac{\text{(3) кгс}}{\text{(4) ед. } c_p} \cdot 10^3 \right]$	$\varkappa_{c_p} = 1{,}5-5 \left[\frac{\text{(3) кгс}}{\text{(4) ед. } c_p} \cdot 10^3 \right]$
(5) Для средней дальности	14—20	9—12
(6) Для большой дальности	110—190	60—90
(7) Сверхзвуковые пассажирские самолеты	250—300	140—150

Key: (1). Designation/purpose of aircraft. (2). For feeder lines. (3). kg. (4). unit. (5). For medium distance. (6). For long range. (7). Supersonic passenger aircraft. (8). Condition.

Section II.

Selection of diagram, power plant and basic parameters of aircraft.

Chapter VI.

Airplane design, analysis and the selection of diagram.

Airplane design is determined by relative position, form and quantity of basic components of aircraft - wing, fuselage, tail assembly and engines.

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By quantity of aggregates/units are distinguished following diagrams: biplane and monoplane (Fig. 6.1a, b); with one or two fuselages (Fig. 6.1c, d); with one or two surfaces of horizontal or vertical tail assembly (Fig. 6.1e); with one or several engines. Are possible airplane designs without the wing (the "flying fuselage") [1], without fuselage (the "flying wing"), and also without the horizontal tail assembly ("bobtailed aircraft").

FOOTNOTE [1]. At the speeds, which correspond to M≥5, the fuselage little is inferior to wing according to the lift effectiveness. At the low speeds the lift can be created by the thrust of special engines. ENDFOOTNOTE.

These diagrams are given in Fig. 6.1f, g, n, o, t.

Form of aggregates/units can change in flight (transformed diagrams, Fig. 6.1h, i, j).

Let us incidentally note that aircraft, planned according to diagram "bobtailed aircraft", in the world was for the first time created in our country (aircraft BICh-3, 1926, designer B. I. Cheranovskiy).

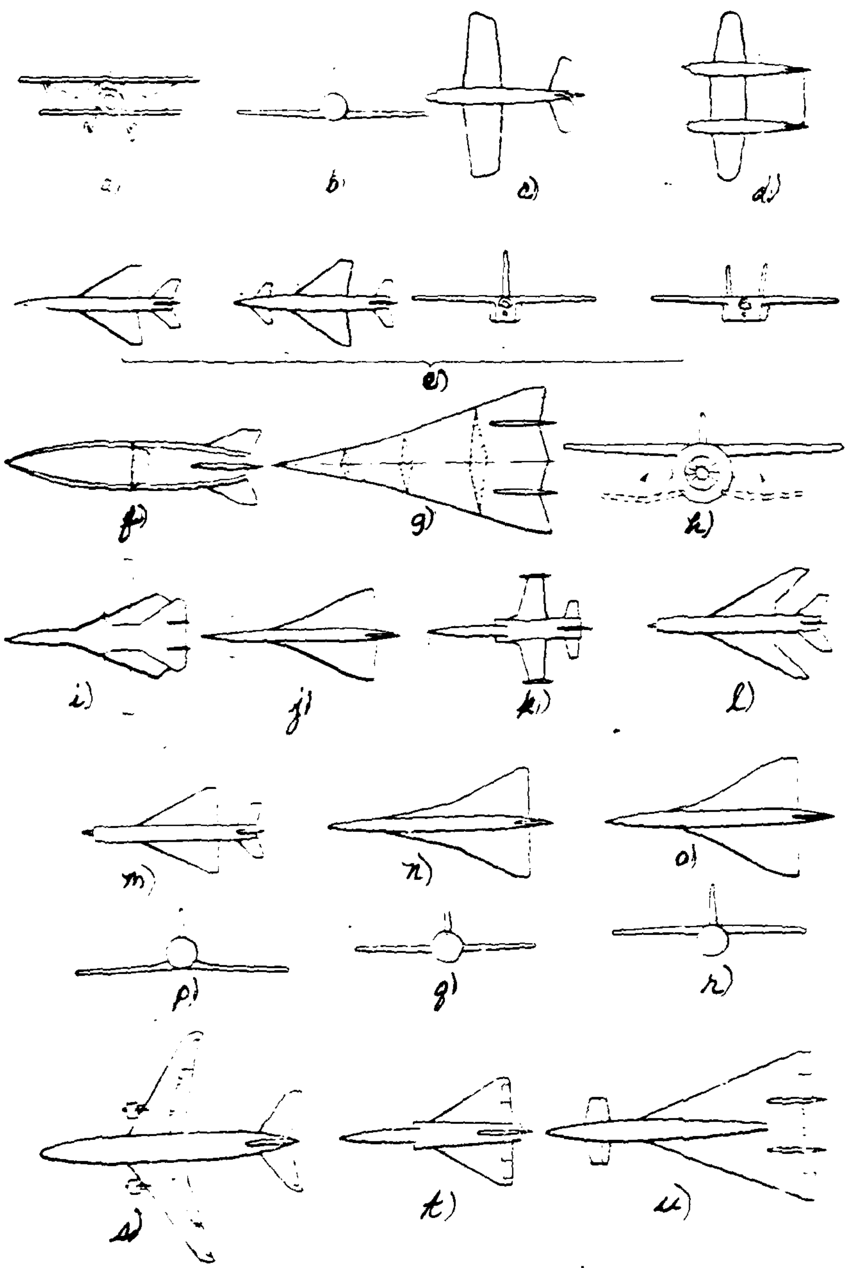

Fig. 6.1. Standard airplane designs.

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Characteristic feature of diagram is created by wing planform -

aircraft with straight/direct, swept, delta wings, with wing of variable on spread/scope of sweepback (Fig. 6.1k-o). The distinctive special feature of diagram create wing arrangement relative to fuselage on the height/altitude (Fig. 6.1p, q, r) - low wing monoplane, midwing monoplane, high wing monoplane.

Most important differences according to aerodynamic, weight and operating characteristics creates mutual wing arrangement and horizontal tail assembly along the length of fuselage - diagram normal, or classical (Fig. 6.1s), "bobtailed aircraft" (Fig. 6.1t) [1] and the diagram "canard" (Fig. 6.1u).

FOOTNOTE [1]. Elevons on the trailing wing edge here perform the role of horizontal tail assembly. ENDFOOTNOTE.

Not less important sign/criterion of diagram is the location of engines on the aircraft (see § 4).

But how to explain this diversity of diagrams? Basic reason - in the diversity of requirements for the aircraft, in the continuous development of the possibilities of their satisfaction. Each of the diagrams reflects the tendency of designer to in the best way satisfy tactical technical requirements in this state of development of aviation science and technology.

This tendency it leads sometimes and to solutions, which do not justify itself in practice. The schematic of the unsymmetric aircraft BV-141 (firm Blohm and Voss), constructed in Germany in 1938 (Fig. 6.2), can serve as an example of this solution.

Idea to radically improve survey/coverage of pilot on single-engine reconnaissance aircraft is interesting in this diagram.

From entire diversity of questions, connected with selection of airplane design, let us consider further following important and actual:

- analysis and selection of schematic of supersonic aircraft;

- analysis and selection of schematic of subsonic aircraft;

- airplane design with wing of variable in flight sweepback;

- location of engines on aircraft.

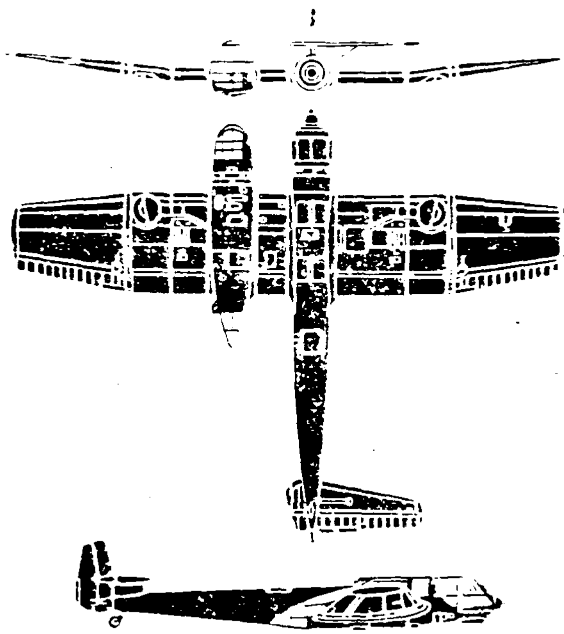

Fig. 6.2. Unsymmetric aircraft BV- 141 of firm Blohm and Voss. Basic data: G_0=4400 kg; $c_{\text{[illegible]}}$=[illegible] kg; S=53.1 m^2; l=17.46 m; [illegible] km/h; [illegible] km/h; motor BMW-80; N_0=1600 l/s.

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Formulation of problem about selection of diagram and parameters of aircraft is examined in § 1 chapter III. Let us recall only that during the assigned operational requirements and other limitations, placed by stress standards, rigidity and for operation, it is necessary among entire diversity of diagrams and parameters of aircraft to select such, which they would lead to the outer limit of the criterion of evaluation of aircraft. As the evaluation criterion are accepted: the efficiency/cost-effectiveness of operation (commercial airplanes), combat efficiency or cost/value of accomplishing operation. The value of takeoff weight is the approximate criterion. The best airplane design of any designation/purpose in the first approximation corresponds to the minimum of takeoff weight [1] during all assigned limitations [4, 25].

FOOTNOTE [1]. During the more detailed comparison should be considered also the effect of diagram on the manufacturability and the maintainability of construction/design, convenience in the operation and the comfort of crew and passengers. ENDFOOTNOTE.

Among many diagrams several interesting, that compete at first are selected/taken (on basis of precomputations and experiment). Then they are in detail investigated quantitatively (on the evaluation criterion) and it is qualitative (according to the signs/criteria, which did not enter the algorithm of optimization).

§ 1. Analysis and the selection of the schematic of supersonic aircraft.

Will consider nonmaneuverable long-range aircraft with fixed/recorded wing [2], and also aircraft of maneuverable class.

FOOTNOTE [2]. Airplane design with the variable sweepback of wing is examined in § 3 of this chapter. ENDFOOTNOTE.

We will consider that the aircraft have a usual running take-off and a running landing [3].

FOOTNOTE [3]. The schematics of VTOL aircraft and STOL are examined in Chapter X, schematics of air space aircraft - in Chapter XI. ENDFOOTNOTE.

During selection of schematic of nonmaneuverable long-range aircraft considerable attention is paid to value of maximum lift-drag ratio K_{max} in cruise, since on it directly depends either flying range with known over-all payload ratio of fuel/propellant ($\bar{G}_T$=const). or value $\bar{G}_T$ and G_0 with L=const.

Maximum lift-drag ratio can be increased, for example, due to decrease of midsections of noncarrying parts, decrease of washed surface of aircraft, and also by reduction in so-called trim drag. The first two methods they do not require special explanations. Let us pause in more detail at the dependence of lift-drag ratio on the trim drag, which depends substantially on the airplane design.

Let us consider classical (normal) diagram and diagram "canard".

From trimmed conditions of aircraft we will obtain:

- for classical diagram (Fig. 6.3a)

$$\frac{Y_{г.о}}{Y_{кр}}=-\frac{x_{д}-x_{т}}{x_{г.о}-x_{т}};$$

- for diagram "canard" (Fig. 6.3b)

$$\frac{Y_{г.о}}{Y_{кр}} = -\frac{x_д - x_т}{x_{г.о} + x_т}.$$

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Assuming that $x_д - x_т = \Delta x$,

$x_{г.о} - x_т = L_{г.о}$ (for the classical diagram);

$x_{г.о} + x_т = L_{г.о}$ (for the diagram "canard"), and carrying values Δx and $L_{г.о}$ and $b_{САХ}$ we will obtain

$$\frac{Y_{г.о}}{Y_{кр}} = \frac{\overline{\Delta x}}{\overline{L}_{г.о}}.$$

At supersonic speeds, as is known, airfoil center of pressure is considerably moved back/ago and $\overline{\Delta}$x increases, and relation $Y_{г.о}/Y_{кр}$ can reach value 0.15-0.20 (in subsonic regime 0.03-0.05). Hence follows that the balancing/trimming in the supersonic regimes leads to a considerable increase in the resistance, connected with the horizontal tail lift.

Total drag of aircraft taking into account balancing/trimming is equal

$$X = X_0 + X_{i\,кр} + X_{бал} = X_0 + X_{i\,кр} + X_{i\,г.о}.$$

After dividing both parts of the equation to qs [we consider that $q_{кр} = q_{г.о}$ and $(c^{\alpha}_{y})_{кр} = (c^{\alpha}_{y})_{г.о}$], we will obtain

$$c_x = c_{x_0} + c_{x_{i\,кр}} + c_{x_{бал}} = c_{x_0} + c_{x_{i\,кр}} + c_{x_{i\,г.о}}\overline{S}_{г.о},$$

or, since for classical diagram $c_y = c_{y_{кр}} + c_{y_{г.о}}\overline{S}_{г.о}$, and for the diagram "canard" $c_y = c_{y_{кр}} - c_{y_{г.о}}\overline{S}_{г.о}$, then respectively

$$c_x = c_{x_0} + D_0(c_{y_{кр}} + c_{y_{г.о}}\overline{S}_{г.о})^2 + D_0 c^2_{y_{г.о}}\overline{S}_{г.о}; \tag{6.1}$$

$$c_x = c_{x_0} + D_0(c_{y_{кр}} - c_{y_{г.о}}\overline{S}_{г.о})^2 + D_0 c^2_{y_{г.о}}\overline{S}_{г.о}, \tag{6.1'}$$

where $D_0 = 1/c^{\alpha}_{y}$.

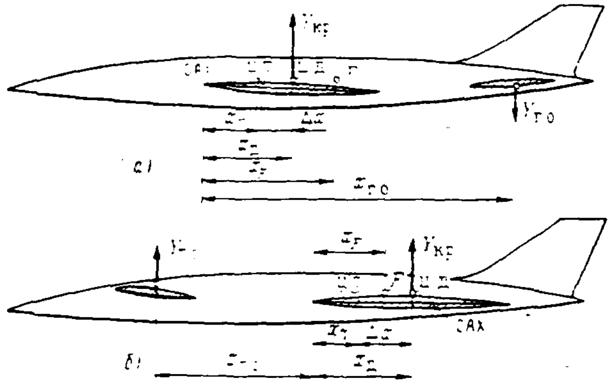

Fig. 6.3. Longitudinal balance of aircraft of classical (normal) diagram (a) and diagram "canard" (b): F - focus of aircraft.

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From Fig. 6.3 it follows that

$$(c_{y_{кр}} + c_{y_{г.о}}\bar{S}_{г.о})\bar{\Delta x} = c_{y_{г.о}}\bar{L}_{г.о}\bar{S}_{г.о} \text{ — для (1) классической схемы;}$$

$$(c_{y_{кр}} - c_{y_{г.о}}\bar{S}_{г.о})\bar{\Delta x} = c_{y_{г.о}}\bar{L}_{г.о}\bar{S}_{г.о} \text{ — для (2) схемы „утка".}$$

Key: (1). for the classical diagram. (2). for diagram "canard".

From these expressions we will obtain

$$c_{y_{г.о}} = \frac{c_{y_{кр}}\bar{\Delta x}}{\bar{S}_{г.о}(\bar{L}_{г.о} \mp \bar{\Delta x})}.$$

For classical diagram in denominator minus is taken, for diagram "canard" - plus.

Let us substitute value $c_{y_{г.о}}$ in (6.1) and (6.1'). After conversion we will obtain for the classical diagram

$$c_x = c_{x_0} + D_0 c_{y_{бал}}^2 \left[1 - \frac{\overline{\Delta x}^2}{(\overline{L}_{г.о} - \overline{\Delta x})^2}\left(1 - \frac{1}{\overline{S}_{г.о}}\right) + \frac{2\overline{\Delta x}}{\overline{L}_{г.о} - \overline{\Delta x}}\right]; \quad (6.2)$$

for the diagram "canard"

$$c_x = c_{x_0} + D_0 c_{y_{бал}}^2 \left[1 + \frac{\overline{\Delta x}^2}{(\overline{L}_{г.о} - \overline{\Delta x})^2}\left(1 - \frac{1}{\overline{S}_{г.о}}\right) - \frac{2\overline{\Delta x}}{\overline{L}_{г.о} - \overline{\Delta x}}\right]. \quad (6.2')$$

Thus, $c_x = c_{x_0} + D_{0\,бал} \cdot c_{y_{бал}}^2$.

where $D_{0\,бал} = (1 + \omega) D_0$,

moreover ω - coefficient, which considers the effect of balancing/trimming on the resistance of aircraft.

For the classical diagram

$$\omega = \frac{\overline{\Delta x}^2}{(\overline{L}_{г.о} - \overline{\Delta x})^2}\left(1 + \frac{1}{\overline{S}_{г.о}}\right) - 2\frac{\overline{\Delta x}}{\overline{L}_{г.о} - \overline{\Delta x}}; \quad (6.3)$$

for the diagram "canard"

$$\omega = \frac{\overline{\Delta x}^2}{(\overline{L}_{г.о} - \overline{\Delta x})^2}\left(1 - \frac{1}{\overline{S}_{г.о}}\right) - 2\frac{\overline{\Delta x}}{\overline{L}_{г.о} - \overline{\Delta x}}. \quad (6.3')$$

Maximum aerodynamic aircraft quality/fineness ratio, as is known, is equal

$$K_{max} = \frac{1}{2}\sqrt{\frac{1}{c_{x_0} D}}.$$

Taking into account the trim drag

$$K_{max} = \frac{1}{2}\sqrt{\frac{1}{c_{x_0} D_{бал}}}.$$

or

$$K_{max} = \frac{1}{2}\sqrt{\frac{1}{c_{x_0} D_0 (1 + \omega)}}.$$

Formulas given above make it possible to do conclusion about considerable effect of value ω on aerodynamics of aircraft. The greater ω, the greater the trim drag and the greater the loss of lift-drag ratio.

During analysis of trim drag it is convenient to utilize concept of focus of aircraft. As is known, $\bar{x}_F - \bar{x}_T = -m_z^{c_y}$, where $m_z^{c_y}$ - derivative of the coefficient of the pitching moment of aircraft on c_y (it stored up longitudinal static stability);

$$\bar{x}_F = x_F / b_{CAX}; \quad \bar{x}_T = x_T / b_{CAX}.$$

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The more it stored up stability $m_z^{c_y}$, the greater and balancing losses of lift-drag ratio, since increases $\bar{x}_F - \bar{x}_T = \overline{\Delta x} = \bar{S}_{r.o} \bar{L}_{r.o} \frac{1}{1 - \bar{S}_{r.o}}$ (here in brackets plus sign it corresponds to normal diagram, and minus - to diagram "canard").

For stabilization of supersonic aircraft in subsonic regime value $m_z^{c_y}$ is chosen in limits of 0.02-0.05, but in supersonic regimes as a result of movement of focus back longitudinal static stability considerably increases (if we do not take special measures, indicated below), especially on aircraft of normal diagram. Aircraft "bobtailed aircraft" occupies a certain mid-position on the greatest shift/shear of focus back/ago upon transfer from M<1 to M>1 (Fig. 6.4). For the aircraft of this diagram with the flat/plane wing (the flat surface of wing chords) $\bar{x}_F \approx 0.35$ and $\bar{x}_T \approx 0.3$ with M<1, and with M>>1 $\bar{x}_F \approx 0.5$. Thus, "bobtailed aircraft" with the flat/plane wing has with M>>1 $m_{z}^{c_y} = -0.15$. if $\bar{x}_T = 0.30$.

Effect $m_z^{c_y}$ on relative maximum lift-drag ratio $\bar{K}_{max} = K_{max}/(K_{max})_{m_z^{c_y}=0}$ of aircraft of two diagrams is shown in Fig. 6.5. These graphs/curves are constructed with typical values $\bar{L}_{r.}$ and $\bar{S}_{r.}$ the aircraft of normal diagram and diagram "canard". It is evident that the aircraft of classical diagram (with the flat/plane wing) has the large losses of maximum lift-drag ratio, than the aircraft of the diagram "canard".

This fact brought recently to thought about advisability of applying diagram "canard" for heavy long-range aircraft (for example, aircraft XB-70 "Valkyrie", USA). Then it was explained that losses K_{max} from the balancing/trimming can be significantly decreased or even eliminated for the diagrams in question, if we use the following constructive solutions:

1) the "floating" or retractable with M<1 horizontal tail assembly in the forepart/nose aircraft component.

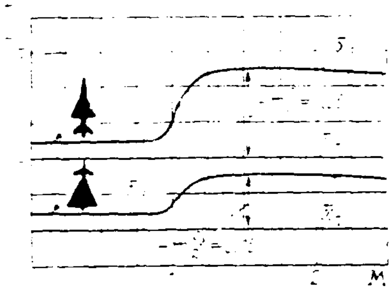

Fig. 6.4.

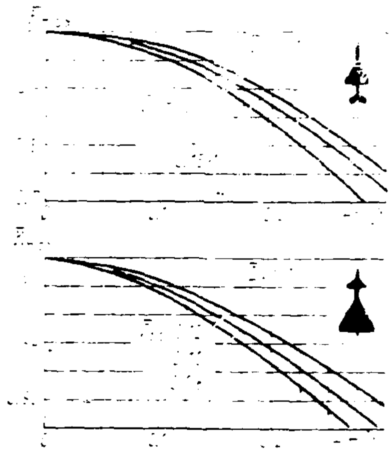

Fig. 6.5.

Fig. 6.4. Change in longitudinal stability factor of aircraft of normal diagram and diagram "canard" (wing flat/plane).

Fig. 6.5. Dependence of relative lift-drag ratio on longitudinal stability factor (wing flat/plane):

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At subsonic speed this tail assembly either is removed in the fuselage or as weathervane, is established/installed at zero or certain constant positive angle of attack and thus does not affect the position of the focus of aircraft. The aircraft of the diagram "canard", for example, with this tail assembly with M<1 and is, actually, "bobtailed aircraft". At supersonic speed the tail assembly is pinched at the specific angle of attack and it displaces the focus of aircraft forward to value $\Delta\bar{x}_F = \bar{S}_{r.o} L_{r.o}$ (here $\bar{S}_{r.o}$ and $L_{r.o}$ relate to the tail assembly - destabilizer). In this case

$m_z^{c_y}$ with M<1 and with M>>1 it can have the identical values of order 0.03-0.05, that also is required (Fig. 6.6).

2) root overflows on the wing (Fig. 6.7) before the center of gravity of aircraft. With M>1 the efficiency of overflows due to increase/growth $c_{y\,\text{нап}}^{\alpha}$ increases. Therefore with an increase in Mach number of flight the focus of aircraft is advanced and with calculated Mach number the longitudinal stability factor decreases to the acceptable sizes/dimensions;

3) the deformation of middle surface of wing (surface of chords). Wing seemingly is adjusted to basic regime (c_y, M) of the flight, during which the losses to the balancing/trimming are reduced to the minimum (Fig. 6.8);

4) the deflected/diverted with M>>1 ends/leads of the swept or delta wing; wing tips are converted into the supplementary fins and do not participate in the lift formation. The focus of aircraft with this operation is shifted/sheared forward to value $\overline{\Delta x}_F = \overline{S}_K \overline{L}_K$, where $\overline{S}_K$ - ratio of the area of the deflected/diverted part of the wing to gross wing area; $\overline{L}_K = L_K / b_{CAX}$; L_K - distance from the center of gravity of aircraft to the center of pressure of the deflected/diverted part of the wing (along the axis of aircraft).

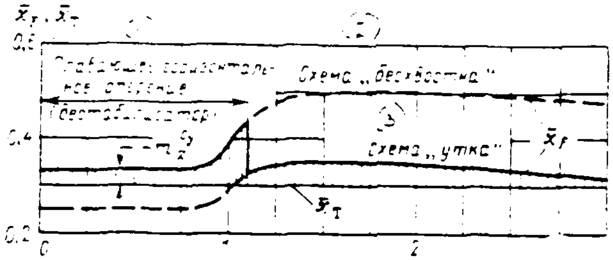

Fig. 6.6. Application of floating horizontal tail assembly on aircraft of diagram "canard".

Key: (1). Floating horizontal tail assembly (destabilizer). (2). Diagram of "tailless aircraft". (3). Diagram "canard".

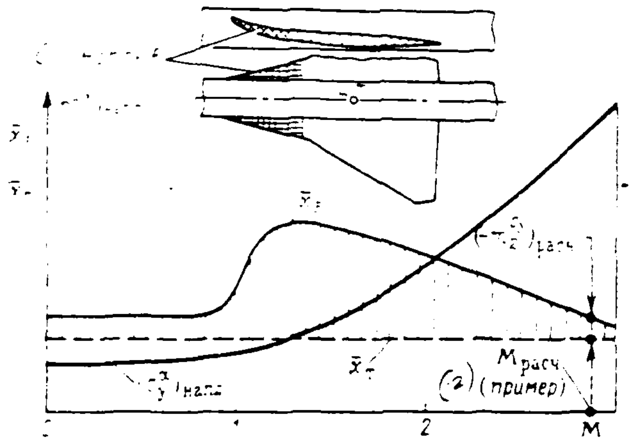

Fig. 6.7. Application of overflow in root of wing.

Key: (1). Overflow. (2). (example).

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All methods examined above of decreasing longitudinal stability factor of aircraft at supersonic speed are based on limitation of displacement of focus with the help of additional increase (creation) in lift before center of gravity of aircraft or with the help of its decrease after center of gravity. Besides these methods, it is possible to utilize one more, different in principle from preceding/previous, led in p. 5;

5) the displacement back/ago of the center of gravity of aircraft (following the displacement of focus) via the pumping of fuel/propellant of the front/leading tanks into the rear balancing tank. This solution is utilized usually in the combination with those enumerated above. Its deficiency consists in the fact that for the rapid pumping of fuel/propellant are required the very powerful (usually - several ten kilowatts) and heavy pumps, and also the fuel lines of large productivity. When after the pumping of fuel/propellant it is back/ago necessary to specially decrease the flight speed to M<1, it is necessary to use the emergency discharge of the rolled fuel/propellant to avoid the pitch instability in the subsonic zone (Fig. 6.9).

Methods presented allow to a greater or lesser extent to solve problem of lift-drag ratio taking into account balancing/trimming. For the diagrams "canard" and "bobtailed aircraft" it is solved more successfully, for the normal diagram - less successfully in connection with the large shift/shear of the focus of aircraft back/ago.

Let us consider now three schematics of nonmaneuverable supersonic aircraft ("canard" and "bobtailed aircraft" and normal) from other points of view.

On safety of operation aircraft of diagram "canard" is inferior to aircraft of other diagrams. The reasons for this are the following.

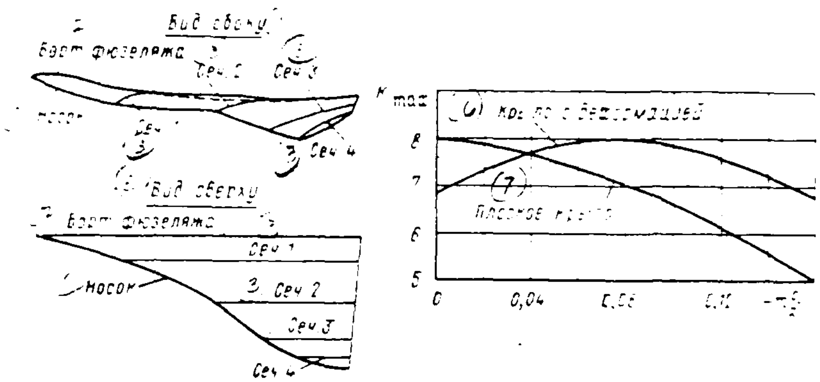

Fig. 6.8. Effect of deformation of wing of aircraft of diagram "bobtailed aircraft" on maximum lift-drag ratio of supersonic aircraft with M=2 (cross section 1-4 and "side of fuselage" at the sight they are on the side center lines of corresponding wing profiles).

Key: (1). Side view. (2). Side of fuselage. (3). Section. (4). Nose/leading edge. (5). Top view. (6). Wing with deformation. (7). Flat/plane wing.

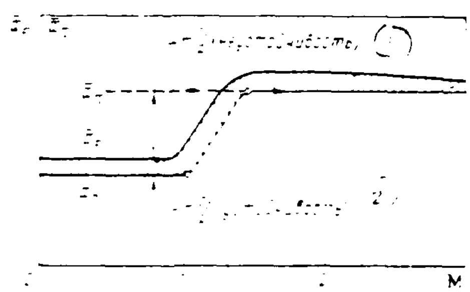

Fig. 6.9. Displacement of center of gravity of aircraft back/ago.

Key: 1). (instability). (2). (stability).

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1. From condition $\quad = \quad$ for diagram "canard" we have:

if $-m_z > \frac{\bar{S}_{t.a.}\bar{L}_{t.a.}}{1-\bar{S}_{t.a.}}$, then $c_{y\,t.a.} > c_{y\,w}$

With typical values $\bar{S}_{t.a.} \approx 0.15$, $\bar{L}_{t.a.} \approx 2$ we have $c_{y\,t.a.} > c_{y\,w}$ if $-m_z > 0.04$. Usually in the diagram "canard" when $M < 1$ $-m_z > 0.04$. Therefore loss of lift at the high angles of attack sets in, as a rule, first on tail assembly ($c_{y\,t.a.} > c_{y\,w}$). In this case $Y_{w} < G$ (at equilibrium $Y_{w} = G - Y_{t.a.}$) and the aircraft of the diagram "canard" together with the peck (moment/torque Y_{w} to the dive relative to the center of gravity of aircraft, see Fig. 6.3) it loses altitude (it sags).

2. Spinning properties of aircraft of diagram "canard" are worse than aircraft of other diagrams.

3. Aircraft of diagram "canard" possesses insufficient dynamic stability (difficult to extinguish short-period oscillations).

It would seem that wing area on this aircraft it is possible to select less than on aircraft of other diagrams, since horizontal tail assembly creates positive lift. However, this advantage the aircraft of the diagram "canard" is deprived. Matter in the fact that the possibility of the high-lift device of wing is here limited by the trimmed conditions and flow separation from the tail assembly. Moreover, the lift of wing as a result of the downwash from the tail assembly decreases by 10-15%. Therefore acceptable takeoff and landing characteristics of the aircraft of the diagram "canard" are achieved by an increase in the wing area [1].

FOOTNOTE [1]. Is possible also the "super-mechanization/super-lift-off device" of horizontal tail assembly (UPS). However, in this case taper after the tail assembly grows/rises, it falls),;. appears stability problem of flow at the engine inlet, etc. ENDFOOTNOTE.

In any case of the gain of area and weight this diagram does not give wing in comparison with the normal diagram.

Aircraft "bobtailed aircraft" (and also "flying wing") has following advantages:

- it is less than loss of lift-drag ratio from balancing/trimming with M>1.2, than in aircraft of normal diagram;

- smaller cost of construction/design (to 10-15%) due to absence of horizontal tail assembly as independent aggregate/unit.

One should note that generally horizontal tail assembly it is necessary to nonmaneuverable supersonic aircraft mainly at high angles of attack (takeoff, landing, output from disruption/separation, etc.). However, in the cruising flight of the function of horizontal tail assembly successfully they can fulfill flaps (in the diagram "bobtailed aircraft" - elevons).

However, during takeoff and landing of diagram "bobtailed aircraft" is inferior to aircraft of normal diagram, since wing of tailless aircraft does not allow/assume mechanization/lift-off device [2].

FOOTNOTE [2]. For the longitudinal balance of aircraft elevons it is necessary to deflect/divert upward, in the direction, opposite to the flap deflection. ENDFOOTNOTE.

For an improvement in the takeoff and landing characteristics of the aircraft of diagram "bobtailed aircraft" ($L_{[illegible]}$, $L_{[illegible]}$, $V_{[illegible]}$, $V_{[illegible]}$) it is necessary to increase wing area (due to low values c_y Fig. 6.10).

It is possible to attain certain positive effect, if we on aircraft of diagram "bobtailed aircraft" use that put forth in nose section during takeoff and landing small wing (so-called "pen", Fig. 6.1j). The area of it does not usually exceed 1.5-2% of the wing area.

During selection of schematic of supersonic nonmaneuverable aircraft sum of weight of wing, fuselage, tail assembly and fuel/propellant acquires important importance.

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On the aircraft of normal diagram it is possible to obtain gain in the weight of wing due to its smaller area (if allows/assumes volume for positioning the fuel), but at the same time it is possible to lose in the weight of fuselage and fuel/propellant (it decreases lift-drag ratio as a result of the losses to the balancing/trimming), and also in the weight of tail assembly.

On aircraft "bobtailed aircraft", vice versa, is obtained smaller fuel load and fuselage (they are absent load on fuselage from horizontal tail assembly), but larger weight of wing.

In each specific case balance of these weights can be either on side of normal diagram or on side of diagram "bobtailed aircraft". Everything depends on the concrete/specific limitations: values of the distance and Mach numbers of flight, length of runway, etc. To give any final recommendations regarding this question is impossible. It is at present clear only that the diagram "canard" on the aircraft of the class in question did not justify those hopes, which on it were laid.

During selection of position of wing relative to fuselage on height/altitude on supersonic nonmaneuverable aircraft one should consider that low wing monoplane has a number of structural/design and operational advantages over high wing monoplane:

- more conveniently to fasten and retract landing gear;

- increases coefficient c_y during takeoff and landing as a result of more effective ground effect ($\Delta c_y \approx 0.1$);

- is ensured buoyancy during ditching.

If conditions for load and unloading (jettisoning) purposeful load do not dictate unambiguously high wing arrangement, then one should approach schematic of low wing monoplane or semi-low-wing monoplane all the more that coefficient c_x of supersonic aircraft - low wing monoplane in practice does not differ from c_x high wing monoplane.

Maneuverable supersonic aircraft [1] must satisfy following basic requirements:

- maximum reserve thrust for guaranteeing of rapid dispersal/acceleration and necessary vertical velocity;

- high-lift device of wing in entire speed range for guaranteeing of rapid turn and change in altitude;

- velocity of landing approach with poor visibility must not require too high a qualification of pilots.

FOOTNOTE [1]. Aircraft with the usual takeoff and the landing and the fixed/recorded geometry of wing here are examined. ENDFOOTNOTE.

During selection of schematic of maneuverable supersonic aircraft should be considered these requirements.

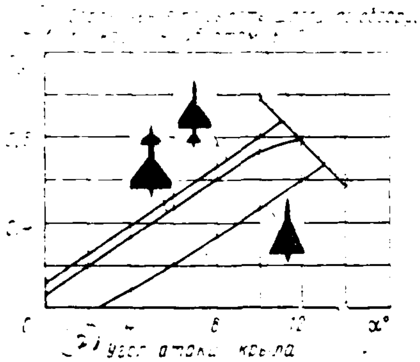

Fig. 6.10.

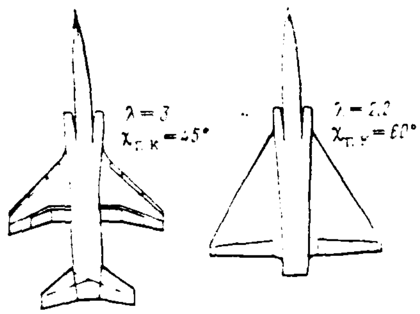

Fig. 6.11.

Fig. 6.10. Lift coefficient during takeoff (without taking into account ground effect).

Key: (1). Limitation on the height of landing gear, on the survey/coverage, on the contact by tail of runway. (2). Angle of attack of wing.

Fig. 6.11. Standard schematics of maneuverable supersonic aircraft (to Table 6.1).

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For comparison let us take two diagrams - normal and "bobtailed aircraft" (Fig. 6.11) with following initial data: calculated number M=2.2; radius of action 550 km; takeoff run length - is not more than 750 m; approach speed is not more than 280 km/h; crew 1 man/person; purposeful load 1500 kg; power plant - one TRD [turbojet engine].

Results of comparison are given in Table 6.1. Table 6.1 shows the advantage of normal diagram by the takeoff weight (in essence, due to the smaller wing area). This advantage is achieved/reached as a result of the fact that the wing of the aircraft of norral diagram allows powerful lift-off device (for example, double-slotted extension flaps and the drooped nose), whereas the wing of "bobtailed aircraft" does not allow/assume mechanization/lift-off device.

Parameter $p_0 c_{y\,доп}$ characteristic radius of turn (here $c_{y\,доп}$ - permitted from condition of stalling or buffeting lift coefficient), is virtually identical with respect to value for both diagrams. The characteristic parameter of dispersal/acceleration - g-force $n_x = \frac{P - X}{G}$ - is more in normal diagram due to the smaller wing area. Calculations show that the time of climb with the simultaneous dispersal/acceleration to $M_{разг}$ ($H = 15$ km; $M_{разг} = 2.2$) aircraft "bobtailed aircraft" has to 25% more than the aircraft of normal diagram.

Due to comparatively light load on m² of wing aircraft "bobtailed aircraft" is more sensitive to vertical gusts of air (more g-force n_y from gusts, than in aircraft of normal diagram), which impedes piloting, tires pilot and decreases lifetime of construction/design (especially in flight at low altitudes).

Advantage of diagram "bobtailed aircraft" - somewhat smaller takeoff run length [1] - cannot change advantage of normal diagram.

FOOTNOTE [1]. During this setting of the problem, where $V_{зл} = const$

ENDFOOTNOTE.

One should stress that obtained conclusions are not consequence of initial data accepted, faster they reflect organic deficiency in "bobtailed aircraft" - unresolved problem of high-lift device of wing. Conclusions are sufficiently general/common for the formulation of the problem accepted about the comparison of the aircraft of different diagrams with the identical degree of the safety of operation, determined to a considerable extent by approach speed.

§ 2. Analysis and the selection of the schematic of subsonic aircraft.

For lifetime of aviation is known considerable attempts to use diagrams "bobtailed aircraft" and "canard" for subsonic aircraft [24]. However, these attempts, as a rule, were finished with the experimental models or the short runs of aircraft.

Table 6.1. Comparison of two schematics of maneuverable supersonic aircraft.

Схема	Параметры											
	G_0 в кгс	S в м²	P_0 в кгс	p_0 в кгс/м²	$\bar{P}_0$	$V_{пр}$ в км/ч	$c_{y\,max}$	$c_{y\,отр}$	$L_{разб}$ в м	n_y (H = 0, M = 0,9)	$p_0/c_{y\,доп}$ (M = 0,9)	$\bar{G}_{пуст}$
Нормальная	[illegible]850	23,5	10 300	585	0,745	26[illegible]	1,35	1,2	750	0,56	585	0,610
«Бесхвостка»	[illegible]4 400	50,5	[illegible]1 500	285	0,8[illegible]	28[illegible]	[illegible]	0,66	610	0,48	570	0,618

Key: (1). Diagram. (2). Parameters. (3). in kg. (4). in m². (5). in kgf/m². (6). in km/h. (7). m. (8). Normal. (9). "Bobtailed aircraft".

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Normal schematic of subsonic aircraft maintained/withstood testing by time and by practice now it is classical.

Basic reasons for failures of aircraft of diagrams "bobtailed aircraft" and "canard" were connected with insufficient stability and controllability, with impossibility or limitedness of high- lift device of wing. Flight safety on such aircraft was poorer, but drag and weight - not are less (due to an increase in the wing area), than in the aircraft of normal diagram.

Let us consider following varieties of normal diagrams:

a) diagram created by wing arrangement on height/altitude of fuselage (see Fig. 6.1p, q, r);

b) diagram with application of twin-boom fuselage instead of usual single-beam (see Fig. 6.15) [1].

FOOTNOTE [1]. The schematics of the arrangement/position of engines on the subsonic aircraft are examined in § 4. ENDFOOTNOTE.

To advantages of diagram with high wing arrangement they relate:

- decrease of aerodynamic drag from interference, especially for circular fuselage (Fig. 6.12);

- decrease of distance from fuselage to earth/ground, which creates series/row of operational conveniences;

- good coverage of ground from passenger compartment;

- reduction in probability of breakdown of engines, arranged/located on wing, as a result of entry/incidence of solid particles with runways during takeoff and landing (Fig. 6.13).

For decreasing harmful wing-root interference effect fillets usually are established/installed. And the nevertheless lift- drag ratio of high wing monoplane with the circular fuselage to 4-5% is more than in low wing monoplane (other conditions being equal). In the case of rectangular cross section fuselage and in the midwing monoplane fillets it is possible not to establish/install.

In spite of aerodynamic advantages of average/mean wing arrangement, this diagram rarely is used for contemporary subsonic aircraft for layout reasons: wing usually is passed to zone of pilot's, passenger or cargo compartment.

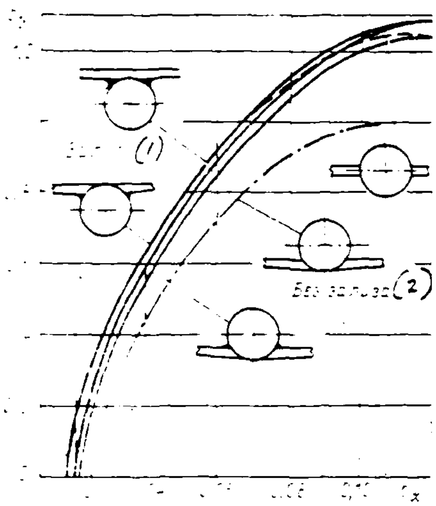

Fig. 6.12.

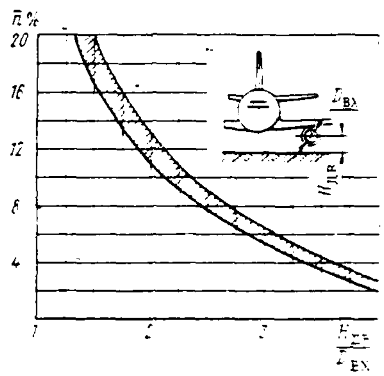

Fig. 6.13.

Fig. 6.12. Polars of aircraft in different position of wing on height/altitude of fuselage.

Key: (1). Fillet. (2). Without fillet.

Fig. 6.13. Percentage of before the appointed time taken/removed engines (TRD) depending on their distance to runways.

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On military transport and cargo aircraft high wing arrangement is most acceptable from operational point of view, it gives possibility to substantially decrease distance from floor of cargo compartment to earth/ground and to facilitate loading and unloading.

Schematic of wing arrangement on height/altitude of fuselage affects, as can be seen from Fig. 6.13, service life of engines, if they are arranged/located on wing. This effect can be approximately considered according to the formula

$$T_{\text{дв}} \approx T_{\text{дв.стенд}} \left(\frac{100 - \bar{n}}{100} \right). \qquad (6.4)$$

where $T_{\text{дв}}$ - average/mean service life of the engines, installed on the aircraft, in the hours;

$T_{\text{дв.стенд}}$ - average lifetime, established/installed on the stand, in the hours;

$\bar{n}$ - is taken from the graph (Fig. 6.13) in terms of the average/mean value of height/altitude $H_{\text{дв}}$ for all engines, established/installed on the aircraft, in percentages.

Calculations according to formula (6.4) show that the service life of engines on the high-wing monoplane can be to 10-15% more than on the low wing monoplane. Therefore the prime cost of transportation, which depends on $T_{\text{дв}}$ (see Chapter 1), on the high-wing monoplane is reduced. However, in the absolute value the efficiency/cost-effectiveness of the operation of high-wing monoplane in the majority of the cases is obtained somewhat worse than low wing monoplane, due to the weight losses. These losses are explained by the following reasons:]

- on the high-wing monoplane it is necessary to specially strengthen the lower part of the fuselage in the case of crash landing without the landing gear;

- increases the weight of the load-bearing elements (frames/formers) of fuselage, which receive loads from the wing and landing gear (if basic landing gear struts are joined to the fuselage);

- by 30-50% increases required fin-and-rudder area in connection with a deterioration in the lateral stability of high-wing monoplane at the high angles of attack, when tail assembly falls into wake from the wing (Fig. 6.14).

In sum weight of structure of high-wing monoplane increases by 2.5-3% of takeoff weight, if all landing gear struts are fastened to fuselage, and to 0.7-1.0%, if basic struts are fastened to wing.

When in the case of emergency splashdown it is necessary to ensure buoyancy (which high-wing monoplane in contrast to low wing monoplane does not possess), then it is necessary to establish/install special pneumatic floats, which also increase weight of high- wing monoplane.

During final solution of question about wing arrangement concerning height/altitude of fuselage it is necessary thus to consider series/row of contradictory factors: high wing monoplane has best aerodynamic and operational characteristics, but it is inferior to low wing monoplane by weight of construction. Therefore it is necessary to make the detailed calculation of weight and efficiency/cost-effectiveness of aircraft of both diagrams. But if the discussion deals with the design of military transport or cargo aircraft, then for layout and operational reasons should be given preference to high wing arrangement.

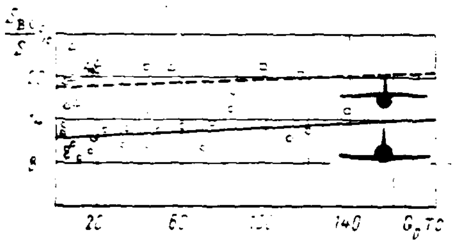

Fig. 6.14. Relative fin-and-rudder area of high-wing monoplane and aircraft-low wing monoplane: O - aircraft-low wing monoplanes; Δ - high-wing monoplanes; ▭ - military transport high-wing monoplanes.

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During design of multipurpose (for national economy) and cargo aircraft need for considering advantages and shortcomings in twin-boom design of fuselage (with nacelle) instead of usual fuselage (Fig. 6.15) appears.

Convenience in load and unloading of nacelle is advantage of twin-boom design. However, the realization of such, is at first glance, tempting diagram it runs into the series/row of the difficulties, connected with a deterioration in aerodynamics and a gain in weight of construction/design.

Calculations show that aerodynamic drag of aircraft with twin-boom fuselage to 10-15% is more than usual due to larger washed surface and oblique airflow of beams by flow from propellers (on aircraft with PD and TVD). If we decrease the washed surface of beams, after reducing the sizes/dimensions of their cross section, then the problem of the rigidity of fastening horizontal tail assembly and weight of beams themselves appears.

From condition of equal sagging/deflection at end of usual and twin-boom fuselage (i.e. with equal flexural rigidity) it is possible to obtain following relationship/ratio in the first approximation,:

$$G_{11} = G_1 \cdot h_1^2/h_{11}^2,$$

where G_1 - weight of tail section (to wing spar) of usual fuselage;

G_{11} - weight of beams;

h_1 - height/altitude of cross section of usual fuselage in bearing edge (in wing spar);

h_{11} - height/altitude of structural section in bearing edge of its on wing.

Passing to weight of entire fuselage, we have:

$$(G_{\phi})_{11} = (G_{\phi})_1 \left[1 + \overline{G}_1 \left(\frac{h_1^2}{h_{11}^2} - 1\right)\right]. \quad (6.5$$

Here $(G_{\phi})_1$ - weight of usual fuselage;

$(G_{\phi})_{11}$ - weight of girder fuselage (together with the nacelle);

$\overline{G}_1$ - ratio of the weight of the tail section of the usual fuselage to its gross weight $\overline{G}_1 = G_1/(G_{\phi})_1$.

In aircraft with weight of up to 6-8 t $\overline{G}_1 = 0.25-0.3$. If, for example, $h_1/h_{11} = 1.5$. then the weight of twin-boom fuselage, as it follows from formula (6.5), there will be to 31-37% greater than usual fuselage (with the equal flexural rigidity).

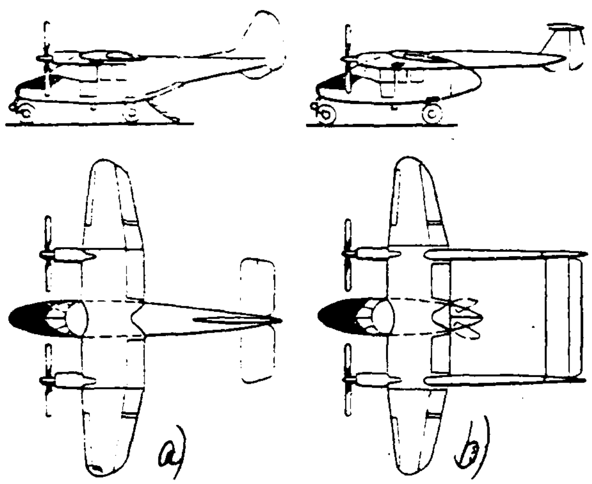

Fig. 6.15. Usual (a) and twin-boom (b) airplane designs with TVD.

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Besides gain in weight and aerodynamic drag, aircraft with twin-boom fuselage has other shortcomings:

- cost/value of fuselage more (to 10-15%) due to special equipment for manufacture and assembly of beams;

- beam/gully occupy part of spread/scope of high-lift device of wing; therefore wing area is necessary to increase by 5-7% for achievement of identical takeoff and landing characteristics with usual aircraft;

- weight of control line grows by 20-25%.

As a result it is possible to do conclusion about inexpediency of application of twin-boom fuselage, on the basis of weight and aerodynamic characteristics.

§ 3. Airplane design with wing of variable in flight sweepback.

Sweepback of wing variable in flight is special case of variable/alternating geometry of aircraft. This idea is not new. Even at the glow of aviation French designer Clement Ader constructed the model airplane (1904), in which changed the sweepback of wing and horizontal tail assembly "for regulating the speed".

Aircraft mono-biplane (see Fig. 6.1h) was constructed by us in 1940 (designers V. V. Nikitin and V. V. Shevchenko). In 1931 near Paris the aircraft of the construction/design of Makhonin with the variable wing area by means of the advancement of arms tested. The wingspan varied from 13 to 21.1 m, wing area increased from 21 m^2 to 33 m^2. The weight of the mechanization/lift-off device, connected with the telescopic separation of this wing, was 850 kg (17% of design take-off weight).

Already in the forties in many countries work on creation of aircraft with wing of sweepback variable in flight was begun.

At present this idea became the same customary as idea of retractable landing gear or extension flaps.

However how to explain that for course of entire history of development of aviation were done attempts to create geometry of aircraft variable in flight? The reason for this phenomenon is included in the contradiction between the invariability of the forms of aircraft and a change in the flight conditions along the speed and the height/altitude. Airplane design with the fixed/recorded geometry is designed for the narrow flight envelope, and output/yield from this range makes aircraft far from optimum.

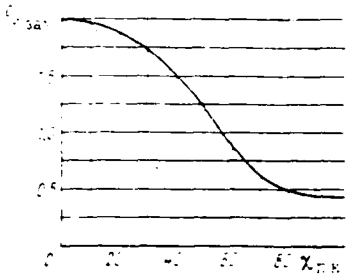

Fig. 6.16.

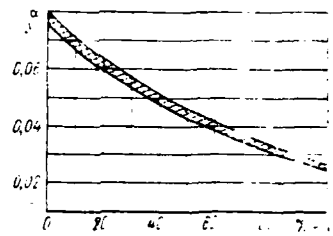

Fig. 6.17.

Fig. 6.16. Dependence $c_{y\,\text{заx}}$ with landing approach on sweepback of wing on leading edge (mechanization/lift-off device: double-slotted flaps and slats; α=12°).

Fig. 6.17. Character of change c_x^0 of aircraft in dependence on sweepback of wing on leading edge (M<1).

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The tendency to adapt the geometry of aircraft toward the changing flight conditions led to the creation of the diagrams of the variable geometry (the "adapted diagrams").

Rapid development of this idea is recently explained by practical requirements for aircraft with very large flight envelope (multimode aircraft).

Aircraft with wing of sweepback variable in flight has following advantages.

1. On supersonic aircraft it is possible to considerably improve

takeoff and landing characteristics due to increase in elongation/aspect ratio and c_y^a, and also effect of high-lift device of wing at minimum sweep angle (Fig. 6.16). Thus, when $\chi_{п.к}=20^\circ$, $\lambda=6-7$, utilized coefficients c_y are 2-2.5 times more than on the aircraft with $\chi_{п.к}=60^\circ$, $\lambda=2.3$. Increase c_y gives the possibility to decrease $V_{отр}$ and $V_{пос}$ by 40-60%, and $L_{разб}$ and $L_{пробег}$ - 2-2.5 times.

2. Lift-drag ratio average/mean during flight substantially grows in connection with increase/growth K_{max} in subsonic regimes

$$K_{ср}=\frac{1}{L}\int_0^L K\,dL \approx K_{M<1}\frac{L_{M<1}}{L}+K_{M>1}\frac{L_{M>1}}{L}. \qquad (6.6)$$

If, for example, on the long-range aircraft with $\chi_{п.к}=\text{var}$, $K_{M<1}=15$, $K_{M>1}=8$, $L_{M<1}/L=0.2$, $L_{M>1}/L=0.8$, then according to formula (6.6) $K_{ср}=9.4$.

On aircraft with fixed/recorded low-aspect-ratio wing when $K_{M<1}=10$, $K_{M>1}=8$ we have under the same conditions $K_{ср}=8.4$, by one it is less.

Increase $K_{ср}$ (when $\chi_{п.к}=30-35^\circ$) gives possibility either to increase flying range with G_0=const or to decrease takeoff weight with L=const due to reduction in fuel load.

3. G-force from vertical gusts of air with M=0.9-1.2 (near earth/ground) can be lowered on $\Delta n_y=1.0-1.5$ because of decrease c_y^a with increase in sweepback of wing (with 45-50° on aircraft with $\chi_{п.к}$=const

to 70- 75° even more in aircraft $\chi_{пр}=\text{var}$. Fig. 6.17).

Let us recall that $\Delta n_{г}=\xi c_{y}^{\alpha}\rho V W_{г}\frac{1}{G/S}$, where ξ=const, $W_{г}$ - speed of gust.

Decrease of g-force from gusts, as it was noted above, favorably affects precision/accuracy of piloting, is decreased enervation of pilot and it contributes to increase in lifetime of construction/design.

4. With increase in sweepback of wing to 75-90° its effective relative thickness is reduced along flow, which leads to certain reduction in coefficient c_{x0}, decrease of booster duration of aircraft and weight of corresponding fuel/propellant.

All enumerated advantages of aircraft with variable sweep wing it is possible to obtain only at a cost of gains in weight of construction/design: due to in area of hinge joint), due to drives and high-lift device of wing. In the sum these weight expenditures compose 3.5- 4.5% of the takeoff weight.

During selection of airplane design with wing of variable sweepback is of interest this diagram when wing in position of greatest sweepback with the help of special flaps completely "it is combined" with horizontal tail assembly, creating with it single smooth profile/airfoil. Aircraft becomes "bobtailed aircraft". In this diagram it is possible to increase the true altitude of profile/airfoil due to the summation of the wing chords and tail assembly ($\bar{c}$=const).

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As a result the structural weight is reduced, especially hinge joint, increases the volume of wing for positioning the fuel/propellant, the resolution of the problem of the guarantee of the required rigidity is simplified.

In conclusion let us give as example results of calculations of supersonic passenger aircraft of four diagrams (Table 6.2), including diagrams with variable sweepback of wing.

From Table 6.2 it is evident that takeoff weight of aircraft from [illegible] is somewhat more than in aircraft of normal diagram. The main reason for this - a gain in weight of the construction of the aircraft with the variable sweep wing.

During comparison of diagrams it was assumed that approach speed must be not more than 275 km/h. In aircraft from χ = [illegible] $=255$ the km/h, since wing area in this case more it was chosen not of the condition of approach speed, but of the condition of positioning/arranging the necessary fuel reserve.

§ 4. Location of engines on the aircraft.

On aircraft of one and the same designation/purpose different diagrams of layout of engines (Fig. 6.18) are used. This attests to the fact that each of the diagrams has advantages, and shortcomings [1].

FOOTNOTE [1]. The diagram, which has only shortcomings, is not competitive. ENDFOOTNOTE.

Let us consider advantages and shortcomings in location on aircraft of most widely used turbojet engines.

During design of maneuverable supersonic aircraft compete two diagrams: combined (single, Fig. 6.18a, b) and separate power plant (Fig. 6.18c, d).

Advantages of combined with fuselage power plant in comparison with separate following:

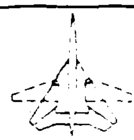
- is less aerodynamic drag (in comparison with diagram d);

- minimum turning moment in the case of failure of one of engines (diagram b);

- increases aspect ratio of high-lift device of wing;

- is improved possibility of suspension of load under wing (in comparison with diagram d).

Table 6.2. Comparison of four schematics of supersonic passenger aircraft $L_{расч}$ = [illegible] км; $M_{расч}$ = 2.7; $G_{ком}$ = [illegible]; $L_{ВПП}$ = 3250 [illegible]

(1) (2)				
(3)	[illegible]	[illegible]	340	340
(4)	[illegible]	[illegible]	[illegible]	[illegible]
(3)	[illegible]	152	[illegible]	137
S (5)	[illegible]	670	[illegible]	730
(6)	520	540	[illegible]	[illegible]
(7)	[illegible]	255	[illegible]	275
(8)	[illegible]	[illegible]	[illegible]	50 [illegible] = 74

Key: (1). Parameters. (2). Diagram. (3). G_0 in tS. (4). G_0 in %. (5). S in m^2. (6). p_0 in kgf/m^2. (7). ... in km/h. (8). ... in deg.

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Shortcomings in combined power plant:

- are more extent and weight of air intakes (due to bending of channels);

- more than loss of velocity head;

- complexity of installation and disassembly of engine and its aggregates/units;

- increases probability of multiple failure of two established/installed in a row engines;

- difficulty of positioning/arranging relatively larger quantity

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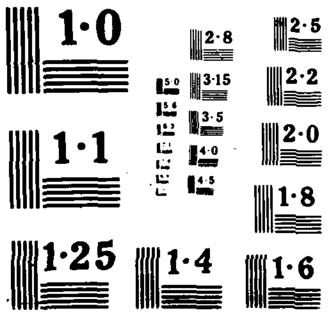
1·0
2·8
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3·5
2·0
1·1
4·0
1·8
1·25
1·4
1·6

of fuel/propellant.

Version c of arrangement/position of engines differ significantly from versions b and d, two given solutions presenting extreme.

Diagram c combines in itself advantages of antipodes b and d:

- comparatively short, simple and light air intake;

- small base drag;

- separate engines;

- favorable possibilities of suspension of load under wing;

- large volume in middle aircraft component for positioning fuel/propellant.

During design of multimode aircraft with wing of variable/alternating sweepback diagram d virtually is excluded due to difficulty to preserve position of axis/axle of engines in process of turn of wing.

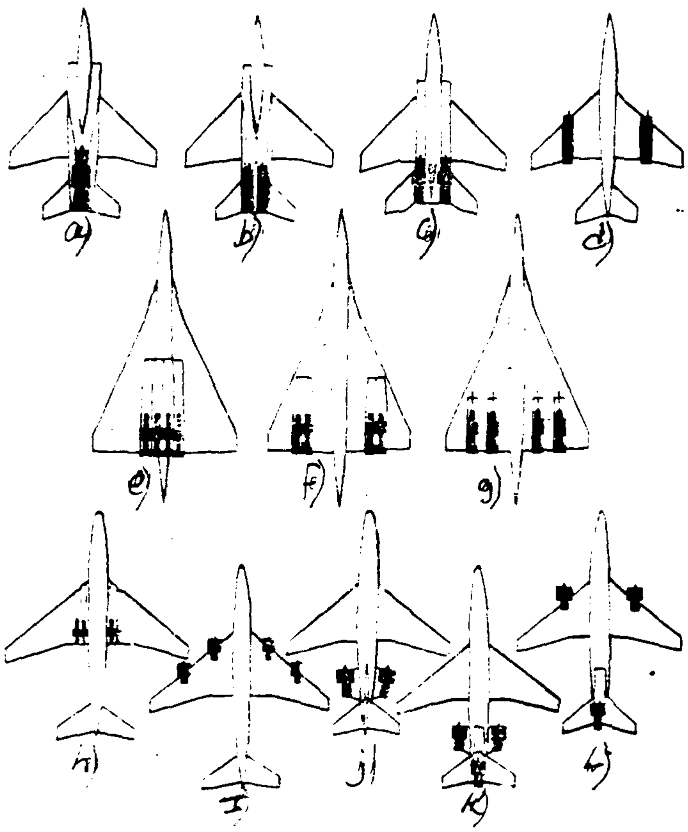

Fig. 6.18. Standard diagrams of layout of engines: a-d - maneuverable supersonic aircraft; e-g - nonmaneuverable supersonic aircraft of long range; h-l - subsonic transport aircraft.

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On nonmaneuverable supersonic long- range aircraft occur the same problems of selection - single or separate power plant (Fig. 6.18e, f, g).

Diagrams e and f are approximately equivalent. On aerodynamics diagram e is somewhat better, but it loses/plays by the weight due to the long air intakes (for an improvement in the entry conditions, in particular for decreasing the boundary layer thickness, air intake it is necessary to lengthen). Diagram e differ significantly from f with respect to the noise, created by engines on the earth/ground during the takeoff (mutual shielding of jets), and also with respect to safety in the case of the failure of engine (less turning moment). However, in diagram f discharging wing from the weight of power plant (somewhat lighter wing) is better. Is important the fact that the sprays and solid particles from the runways from the wheels of leading gear can not reach air intakes in diagram f (angle of sprays from the axis/axle of aircraft it is taken as the equal to 15°).

Diagram g has following advantages:

- short and light air intakes;

- failure of one of engines does not affect work of adjacent engine (since engines are not connected with single air intake as in diagrams e and f);

- simpler maintenance/servicing and replacement of engines.

Shortcoming in diagram g - smaller increase in lift "from compression", created on lower wing surface by shock wave from air intakes (positive interference). In diagrams e and f the effect of an increase of the lift "from the compression" $(+\Delta c_y)$ composes ~20%, while in diagram g approximately half (proportional to wing area, which is located under the action of shock waves). To at the same time place engines in diagram g nearer to the leading wing edge (for an increase in the fields of compression under the wing) is impossible due to the adverse effect of jet on the wing construction (thermal and sonic effect of jet causes the decrease of the service life of construction/design).

Approximate solution of task about optimum selection of schematic of arrangement/position of engines taking into account different contradictory factors can be obtained by method of gradients of takeoff weight, presented in Chapter V.

Final sitting of engines can be done only after careful and detailed study of diverse variants taking into account model tests, in which is imitated work of engines. The typical patterns of the location of engines on the subsonic aircraft are shown in Fig. 18.6(h-l).

Arrangement/position of engines in root of wing (diagram h) extensively was used on heavy subsonic jet aircraft of military and civilian designation/purpose (aircraft Tu-16, Tu-104, Tu-124, etc.; English aircraft "Vulcan", "Victor", "Comet", etc.). This diagram of installation of engines together with the positive qualities (failure of one or two engines on one side does not cause the sharp turning and heeling moments, the high location of air intakes, small external aerodynamic drag) has a number of essential shortcomings (especially for the passenger aircraft).

To them relate:

a) the proximity of exhaust jet to the fuselage covering, strong noise in passenger compartment;

b) long air intakes by 5-6% reduce the engine thrust;

c) the fire, which arose on the engines, can be spread to passenger compartment and fuel tanks (it is required the intensive fire-fighting protection);

d) in the case of the decomposition of compressor blades or turbine possibly the damage/defeat of passenger compartment and fuel tanks (it is required special armoring);

e) the presence of air intakes on the leading wing edge and exhaust ducts on the trailing edge it reduces the possibilities of the high-lift device of wing;

f) hinders the creation of reversing gears of thrust;

g) hinders approach to the engines;

h) substantially is reduced the volume of wing for positioning the fuel/propellant.

Enumerated shortcomings led to the fact that engine installation in root of wing now is not used.

Arrangement/position of engines on pylons under wing (diagram i) widespread on subsonic aircraft. This diagram of installation of engines has the following advantages:

- engines unload wing construction in flight, reducing bending and turning the moments/torques from the external loads, which leads to the reduction of the weight of wing to 10-15%;

- engines damp the oscillations/vibrations of wing in flight in the turbulent atmosphere;

- engines are anti-flutter balancers;

- convenience in the replacement of one type of engine by others (with larger sizes/dimensions);

- light access to the engine during the maintenance/servicing.

Arrangement/position of engines on pylons has shortcomings:

- in case of failure of engine, especially external, is created large turning/running up horizontal moment;

- so that during landing with bank (to 4°) outboard engines they would not concern earth/ground, is required compiling transverse angle of V wing (2-3°), which worsens/impairs stability characteristics and aircraft handling with sweptback wing;

- during low location of engines relative to surface of airfield is possible entry/incidence into air intakes of sand, dust and fine/small stones, which affects service life of engines [see formula (6.4)]. For eliminating this it is necessary to use special measures, for example, the cutoff of the vertical airflow, which are built up from the earth/ground to the air intakes, by air jet, selected/taken from the compressor of engine, which is connected with a gain in weight and the decrease of the engine thrust;

- pylon engine mount impedes the use of flaps throughout entire wingspan, since during the takeoff exhaust jets of engines can break them $(-\Delta c_{y_{cr\,r}}=0{,}08-0{,}12)$.

Arrangement/position of engines on aft fuselage section was for the first time used by firm Sud-Aviation (France) on aircraft of "Caravelle". This diagram of installation of the engines (diagrams j and k) received wide acceptance on the Soviet and foreign passenger aircraft (Il-62, VC-10, Yak-40, Tu-154, etc.).

Arrangement/position of engines on aft fuselage section makes it possible:

- to ensure aerodynamically clean wing, which raises aerodynamic aircraft quality/fineness ratio by 6-9%;

- to maximally use wingspan for positioning/arranging means of mechanization/lift-off device (flaps, slats, etc.), which improves takeoff and landing characteristics of aircraft;

- to determine dihedral of wing from conditions of guaranteeing optimum characteristics of transverse and directional stability and controllability;

- to decrease turning moment with stop of one of engines.

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In addition to this, diagrams j and k make it possible:

- to improve comfort of passengers due to decrease of noise, since engine nacelles in this case are established/installed behind pressurized cabin;

- to raise fire safety, since engines are distant from passenger compartment and from fuel tanks (flame from engine inflaming in flight goes away back/ago, clearing any load-bearing structural elements of aircraft);

- to raise (in comparison with engine installation radically of wing) operating characteristics of power plant and entire aircraft as a whole due to sufficiently good conditions for approach to engines;

- to protect engines from entry/incidence in them of extraneous objects because of high location of air intakes from earth/ground (it increases service life of engines);

- to create best conditions for crash landing of aircraft.

However, diagram of installation of engines on aft fuselage section has essential shortcomings, connected with gain in weight of construction of aircraft due to:

a) intensification of construction/design of aft fuselage section due to further mass and inertia loads of engines (weight of fuselage construction it increases approximately to 10-15%);

b) gain in weight of wing (approximately to 10-15%) due to absence of discharging wing with engines;

c) increase in length of fuselage due to need for fastening engines.

Furthermore, in this diagram centers of gravity of empty and loaded aircraft substantially do not coincide, in consequence of which, difficulties of layout (it is required either fourth support a rest or large and heavy horizontal tail assembly for breakaway during takeoff), appear, appears also need for laying fuel lines from tanks to engines near passenger compartment, which causes danger of entry/incidence of vapors of kerosene into cabin/compartment and increases weight of conduits/manifolds.

One should note that with bypass ratio of engines it is more than 3.5-4.0, when diameter of fan substantially grows, installation of four engines according to diagram j becomes extremely difficult. In this case it is better to use either diagram k or compound configuration l. Latter/last diagram combines in itself the advantages of the wingtip and feed engine installation. The difficulty of the modification of the power plant is a shortcoming in two latter/last diagrams: for the engine with larger diameter the alteration of aft fuselage section together with the air intake is required.

Table 6.3 gives example of comparison by takeoff weight and according to efficiency/cost- effectiveness of operation of hypothetical aircraft in 300 passenger places with three versions of location of engines. Practical flying range with the greatest payload it was assumed to be the equal to 3000 km. Comparison was conducted when $(H, V, L_{\text{ВПП}}, V_{\text{зап}}) = \text{const}$.

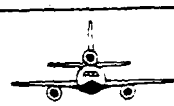
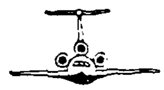
From Table 6.3 it is evident that by takeoff weight and according to efficiency/cost-effectiveness of transportation aircraft with engines, arranged/located according to diagram l, has advantage in comparison with diagrams i and k. Advantage is obtained, in essence, due to the smaller over-all payload ratio of power plant [1] in comparison with diagram i and the smaller structural weight in comparison with diagram k [2].

FOOTNOTE [1]. Thrust-weight ratio is less of the failure condition of one engine; therefore is less the weight of power plant.
[2]. In diagram l wing is unloaded by weight of engines, and fuselage tests/experiences smaller loads from one engine. ENDFOOTNOTE.

In conclusion one should stress that selection of airplane design is complicated creative process, in which more than is somewhere revealed all knowledge, experience and ability of designer.

Table 6.3. Comparison of three diagrams of the layout of engines

$L_{p.a.}$ = [illegible] km/h; $L_{BПП}$ = 2600 m, V = 240 km/h.

(1) \ (2)			
(3)	[illegible]	[illegible]	[illegible]
(4)	[illegible]	100	[illegible]
(5)	[illegible]	[illegible]	[illegible]
(4)	[illegible]	100	[illegible]

Key: (1). Parameters. (2). Diagrams. (3). G_0 in t. (4). G_0 in %. (5). a in kopecks t-km.

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Chapter VII.

BASIC QUESTIONS OF THE DESIGN OF THE POWER PLANT OF AIRCRAFT.

Into power plant of aircraft enter:

1) engines with aggregates/units and systems;

2) air intakes;

3) fuel system [1].

FOOTNOTE [1]. The detailed enumeration of the systems, entering the power plant of aircraft, is indicated in Appendix I. ENDFOOTNOTE.

Theory and design of systems of power plant (engines, air intakes, nozzle, fuel systems, etc.) in detail are illuminated in special literature. In this chapter questions of the design of the power plant, directly connected with the general/common design of aircraft, are examined in essence.

§ 1. Selection of engine for the power plant of aircraft.

For power plant of contemporary aircraft are used jet engines (VRD), piston engines (PD) and liquid-propellant rocket engines (ЖРД [liquid propellant rocket engine]). Jet engines in turn are divided into the gas-turbine (GTD [gas-turbine engine]) and the direct-flow/ramjet (PVRD [ramjet engine]).

GTD at present have widest use in aviation. This class of aircraft engines includes:

- turbojet engines (TRD [turbojet engine]);

- turbojet engines with afterburner (TRDF [turbojet engine with afterburner]);

- turbofan engines (TRDD [turbofan engine]), these engines are called also turbofan (TVRD);

- turboprop engines (TVD).

Piston engines are now placed only on very light aircraft.

Liquid-propellant reaction engines are used in aviation exclusively on experimental aircraft.

Selection of engine for power plant is produced in period of sketch design of aircraft. To consider suitability of one or the other type of engine for the projected/designed aircraft is possible, examining a change of the fundamental engine characteristics in the prescribed/assigned speed range and flight altitudes.

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Chara teristics of aircraft engines.

Fundamental characteristics, on which is accomplished/realized comparative evaluation during selection of engine, are: altitude-speed characteristics P=f(M, H) and $c_r=f(M, H)$, flow rate per second of air $G_в$, weight per horsepower $\gamma_{дв}$ and overall dimensions (length and maximum/overall diameter) of engine [1].

FOOTNOTE [1]. The characteristics of aircraft engines, are necessary for accomplishing the diploma project, they are given in appendix III. ENDFOOTNOTE.

First TRD (with centrifugal compressor) had boost for launching (with V=0 and H=0) less than 2700 kg and sufficiently high specific fuel consumption. At present aircraft engines are capable of developing boost for launching to 30000 kg (GE 4/J5). It does not follow, certain, to assume that TRD with the small thrust gradually died off. Contemporary aircraft engines depending on designation/purpose can have very small thrust, for example TRDD AI-25, established/installed on the aircraft Yak-40, it develops thrust 1500 kg, and thrust TRD Bristol Siddeley BS-347 is equal only 63.6 kg (weight of engine 1.36 kg).

Contemporary GTD depending on purpose can have boost for launching, equal to $P_{0\,max}\approx 50$ kg - smallest full thrust; $P_{0\,max}\approx 30\,000$ kg - greatest full thrust.

Simultaneously with increase in starting specific consumption of fuel GTD was reduced. Up to 1946 the majority of TRD with the single-stage centrifugal compressor had a compression ratio, equal to approximately four, and therefore the specific consumption of fuel of these TRD (in the nonafterburning regime) exceeded 1.3 kg/kg·h.

Contemporary GTD have starting specific fuel consumption, which is changed in the range

$c_p = 0{,}5—0{,}7$ kg/kg·h - nonafterburning regime;

$c_F = 1{,}7—2{,}0$ kg/kg·h - afterburning regime.

It is possible to assume that as a result of further improvements of TRDD specific fuel consumption of less than 0.5 kg/kg·h will be achieved/reached, and on TRDD with high bypass ratio (m=5-8) value of starting specific fuel consumption will approach 0.3 kg/kg·h.

Important parameter, which characterizes perfection of engine, is its specific weight/gravity on boost for launching, i.e., ratio of dry weight of power plant to maximum boost for launching $\gamma_{дв} = G_{дв}/P_0$.

Contemporary GTD can have specific gravity $\gamma_{дв} = 0{,}15—0{,}23$.

One should note that specific weight/gravity of contemporary TRD virtually does not depend on value of boost for launching, for example, TRD GE 4/J5 ($P_0=28\,700$ kg) has $\gamma_{дв}=0{,}166$. TRD J85=13 (8 kg) has $\gamma_{дв}=0{,}163$. Specific weight/gravity of TRDD is proportional to value $\sim P_0^{0.15}$, and also it depends on bypass ratio.

Sometimes instead of weight per horsepower reciprocal value (i.e. thrust-to-weight ratio), which is called weight specific thrust of engine $P_{уд}=1/\gamma_{дв}$ is examined.

Its length and maximum/overall diameter (on compressor, on entry, on nozzle, etc.) are basic size dimensions of engine. The diameter of engine frequently determines the frontal area of fuselage or nacelle, which in the final analysis is reflected in the lift-drag ratio of entire aircraft. Therefore, other conditions being equal, preference is always given to engine with the smaller diameter.

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The dependence of the diameter of engine on the boost for launching can be registered in the form

$$D_{\text{в}} \approx (1.62+0.275m^{\cdots})\sqrt{\frac{P_0}{20\,000}}\ \text{м}. \qquad (7.1)$$

where P_0 - in kg; m - bypass ratio.

Tasks, implemented by contemporary aviation, require of power plant of aircraft of high efficiency/cost-effectiveness over a wide range of velocities and flight altitudes. However, the greatest efficiency/cost-effectiveness of aircraft engine is ensured in such a case, when it is designed for a comparatively small altitude-speed range. By this is explained the division of engines into the subsonic, the supersonic and the hypersonic.

Engines for the subsonic aircraft.

Piston engine is oldest type of engines for subsonic aircraft, as is known. Only low-power piston engines ($N=100-340$ hp) at present are produced and they are established/installed on the very light aircraft (tourist, sport, etc.). However, on the light aircraft PD began to be replaced by other types of engines.

Continuous increase in launching weight of aircraft requires appropriate increase in installed power. Fig. 7.1a gives the curve of an increase in the required power in the cruise setting of flight, beginning from the aircraft of the 40's. Within the period in question the required power increased approximately 20 times. Weight per horsepower according to the thrust horsepower in the cruise setting considerably decreased (Fig. 7.1b), moreover abruptly - with the advent of GTD. Contemporary TRDD develops thrust horsepower on 1 kg of weight 3-4 times more than PD.

As can be seen from Fig. 7.1c, modern GTD (at doubly larger cruising flight speed) in value of specific fuel consumption were equaled with best PD.

Fig. 7.1d shows change in initial velocity of engine installations on 1 hp of thrust horsepower. With the introduction of GTD the cost/value of engine installations sharply was reduced, in spite of a considerable increase in the power of engines.

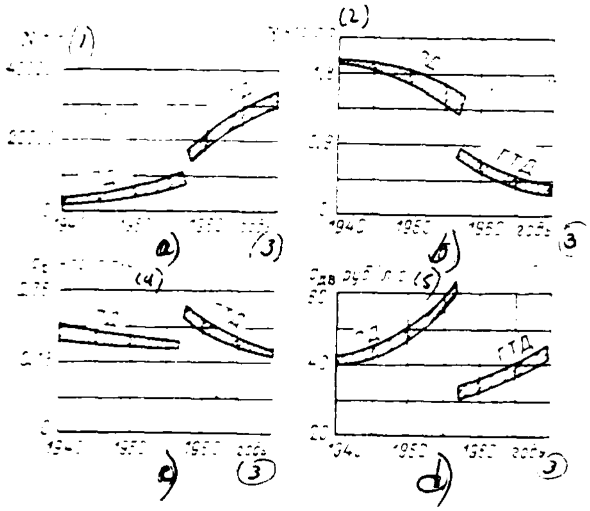

Fig. 7.1. Development of subsonic aircraft engines during period of 1940-1970.

Key: (1). hp. (2). kg/hp. (3). years. (4). kgf/hp·h. (5). rub/hp.

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At present turboprop engine finds increasingly smaller use in aviation. In order to compete with the contemporary TRDD, further improvement of TVD must be directed toward an increase in the specific power and a decrease of the fuel consumption. However, in this case the difficulties, connected with the propeller, appear; it is difficult to create the screw/propeller, which has sufficiently high efficiency at the increased power of engine, but is not less difficult to solve the problem of lowering the general/common noise level and vibration on the aircraft during the installation of a similar screw/propeller.

TRDD (turbofan engine) now is most promising engine for power plant of subsonic aircraft.

TRDD makes it possible to have in cruise to 10-15% smaller specific fuel consumption, than straight jet (Fig. 7.2).

Advantage of TRDD is also higher ratio of takeoff thrust to cruising, consequently, with cruising thrusts (determined by gross weight and aerodynamic aircraft quality/fineness ratio) equal with TRD, TRDD ensures best takeoff data to aircraft.

Service life of this type of engines at present is most high. Service life between the sortings/partitions for TRDD is equal to 8000-10000 h.

First TRDD were worked out on base of existing TRD and they had small bypass ratio m=0.6-1.4 [1].

FOOTNOTE [1]. By degree (coefficient) of bypass configuration, as is known, is understood the ratio of the flow rate per second of air in the outer duct to the flow rate per second of air in the internal duct/contour. ENDFOOTNOTE.

Contemporary TRDD are characterized by higher bypass ratio m=3-6. They have a series/row of fundamentally new constructive solutions (possibility of the independent rpm control of each cascade/stage of compressor); in the construction/design of engines extensively are used new structural materials (in particular, they are used the plastics, reinforced by the filaments of graphite, boron, etc.). These and other innovations make it possible to bring compression ratio in the compressor to 25 and more, which raises the efficiency/cost-effectiveness of engine.

Efficiency/cost-effectiveness of TRDD and its fundamental characteristics depend also on bypass ratio of engine.

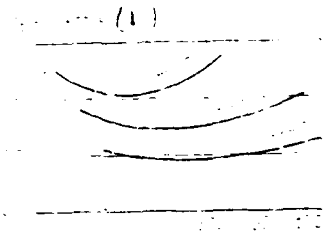

Fig. 7.2.

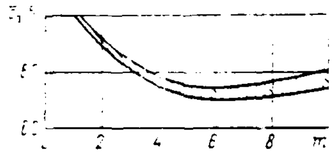

Fig. 7.3.

Fig. 7.2. Typical dependence between temperature of gas before turbine and specific consumption of fuel of TRD and TRDD (under conditions for cruise): m - bypass ratio.

Key: (1). kg/kg•h.

Fig. 7.3. Effect of bypass ratio on value $\bar{c}_T$ TRDD ($\bar{c}_T$ - relative expenditures for fuel/propellant).

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The specific hourly consumption of fuel/propellant in the first approximation, can be expressed as follows:

$$c_p \approx 0.9\left[\frac{0.82}{1-0.525\sqrt[3]{m}} + M(0.494 - 0.0145H)\right] \text{кгс/кгс·ч.} \quad (1) \qquad (7.2)$$

Key: (1). kg/kg•h.

where m - bypass ratio;

M - Mach number of flight;

H - flight altitude, km.

Specific weight/gravity of TRDD approximately can be registered in the form

$$\ldots \approx 0.23 - 0.03m + 0.0052m^{2}. \qquad (7.3)$$

The diameter of engine will be determined according to expression (7.1).

Drag coefficient of engine nacelle of TRDD with increase in bypass ratio is reduced, which is evident from expression

$$c_{x_{м.г}} \approx \frac{0.16}{\sqrt{m}}. \qquad (7.4)$$

Fig. 7.3 shows effect of bypass ratio to efficiency/cost-effectiveness of TRDD (as one it is accepted efficiency/cost- effectiveness of TRDD with m=1-1.5).

At present for heavy subsonic aircraft TRDD with bypass ratio m=4-8 are used.

Subsonic engines include also special lifting TRD and TRDD for VTOL aircraft. According to the principle of the creation of vertical and horizontal thrusts the power plants of VTOL aircraft are divided into the single and the composite/compound. Single power plants serve for vertical takeoff and landing and for the level flight (one and the same engine creates vertical, and horizontal thrusts). These engines, as a rule, are intended for the installation on the supersonic aircraft.

Composite/compound power plants of VTOL aircraft have engines for creation of vertical thrust on takeoff and landing (hoisting) and engines for obtaining horizontal thrust (sustainer). As the sustainer engines are used usual TRD and TRDD.

Lifting engines in parameters of working process and construction/design considerably differ from sustainer engines. These engines have specific weight/gravity approximately 3-3.5 times less than the specific weight/gravity of usual GTD (which is reached, first of all, due to considerable reduction in the service life of engine).

TRDD in comparison with lifting TRD are advantages of lifting;

- smaller exhaust gas velocities (and consequently, smaller destructive action of gas jet on launching pad);

- best efficiency/cost- effectiveness under conditions of vertical takeoff, landing and hovering (in lifting TRDD with low-pressure fan specific consumption of fuel is almost three times less than in lifting TRD). True, this TRDD, possessing the advantages indicated in comparison with lifting TRD, is inferior to it by the diameter and the blockaded volume.

Engines for the supersonic aircraft.

Most economical during supersonic flight are two types of engines - TRD and TRDD.

In order to ensure smallest fuel consumption and highest efficiency over a wide range of Mach numbers (from takeoff to maximum velocity), TRD has, as a rule, afterburner, and TRDD - afterburning i. secondary circuit (Fig. 7.4).

For endurance flight in supersonic regime are examined usually two speed ranges: speed range, which correspond to number M≈2, whose advantage consists in the fact that airframe of aircraft can have construction/design from usual aluminum alloys, and speed range, which correspond to number M≈3, which although creates specific problems, promises higher efficiency/cost-effectiveness of flight.

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One should consider that for flight speed, which corresponds to number M≈2, it is possible to use TRD and TRDD; compression ratio in compressor for both types of engines it must be $\pi_k \approx 9-10$.

Absence of essential difference in characteristics of TRD and TRDD on supersonic speeds to a certain degree is explained by the fact that with increase in coefficient of bypass configuration from 0 to 0.5- 0.7 specific fuel consumption is reduced insignificantly (approximately to 1%). Further increase in the coefficient of bypass configuration progressively worsens/impairs the engine characteristics. One should, however, remember that in this case the discussion deals with the prolonged supersonic flight.

Some supersonic aircraft (multipurpose fighters, etc.) frequently accomplish prolonged subsonic flight. In this case all advantages are on the side of TRDD. Degree of bypass of such engines (in order not to worsen/impair characteristics with M>1) must be small: m=0.7-1.2, while compression ratio - $\pi_k=12-16$

At flight speed, which corresponds to number M≈3, single-circuit and bypass engines also do not have large differences. Therefore in this case it is possible to use engines of both types. However, the difference in the fact that at this velocity engines (both TRD and TRDD) with the low compression ratio in the compressor ($\pi_k=3-4$ when $M\approx3$) have smaller specific consumption of fuel (Fig. 7.5).

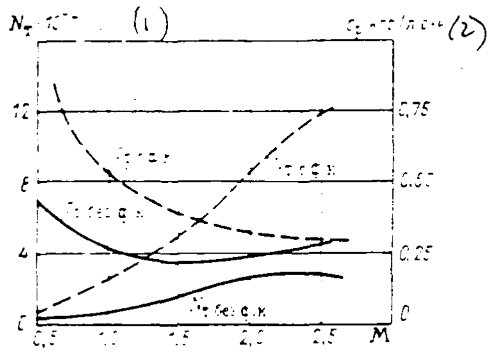

Fig. 7.4.

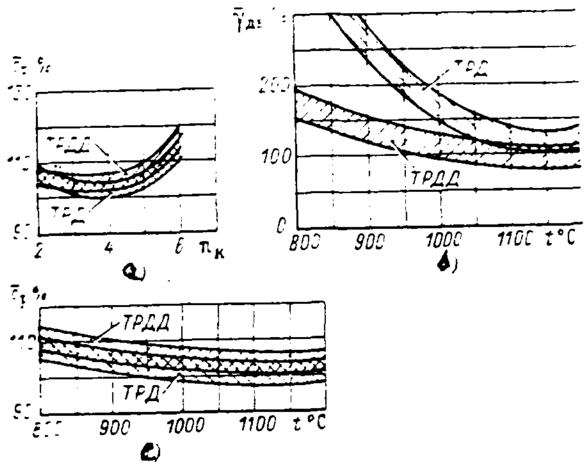

Fig. 7.5.

Fig. 7.4. Dependence of thrust horsepower and specific consumption of fuel of TRD "Olympus" (England-France) on Mach number of flight (with afterburner and without afterburner), H=11 km.

Key: (1). hp. (2). kgf/hp·h.

Fig. 7.5. Characteristics of TRD and TRDD, which determine their efficiency/cost-effectiveness with M=3; H=20 km.

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Engines for the hypersonic aircraft.

Application of gas turbine engines for aircraft power plants is limited, as is known, to number M≈3.5. For the flight at higher velocities the best efficiency/cost-effectiveness will possess PVRD with subsonic combustion (M≤8) and PVRD with supersonic combustion (M≥8).

Rocket engines, which work on chemical fuel/propellant (ZhRD), achieved now this level of development, when further increase in specific impulse becomes ever slower and more expensive. Moreover, the specific impulse of ZhRD remains insufficient for its installation on the hypersonic aircraft (Fig. 7.6).

Efficiency/cost-effectiveness of engine in the first approximation, can be considered in value of its specific impulse J_T (since $c_p = 3600/J_T$ kg/kg·h).

It must be noted that, in spite of increase of specific fuel consumption per supersonic speeds, complete efficiency. engine it increases. Complete efficiency, as is known, considers all losses in the process of the energy conversion of fuel/propellant into the useful thrust work. For the hydrocarbon fuel (kerosene) the expression the complete efficiency of jet engine can be registered in the form

$$\eta_{п} = 0.00082V/c_p. \qquad (7.5)$$

Complete efficiency of engine of subsonic aircraft (M=0.85) composes 24% approximately. At the flight speed, which corresponds to number M=2, it grows already to ~38% (which exceeds the thermal efficiency of the best contemporary power stations), while when $M=3$ $\eta_п \approx 46\%$. Increase/growth efficiency is continued also in flight at the hypersonic speeds with PVRD.

Most promising for hypersonic aircraft are compound engines - direct-flow turbine (TRD+PVRD). TRD must work to the velocity, which corresponds to number M≈3.5, then gas-turbine channel is closed, and at the hypersonic speeds engine will work as PVRD.

Fundamental characteristics examined above of aircraft engines (different types) can have specific differences, caused by design features of concrete/specific samples. For example, two same-type engines with the identical boost for launching can have the different values of specific weight/gravity, specific consumption of fuel, diameter of engine, etc. If all characteristics in one engine are better, a question of the selection of engine is solved unambiguously. However, in practice this case is encountered rarely. As a rule, during the comparison of several engines it proves to be that some characteristics are better in one engine, others - are better in another engine, etc. But how in that case to select concrete/specific engine for the projected/designed aircraft?

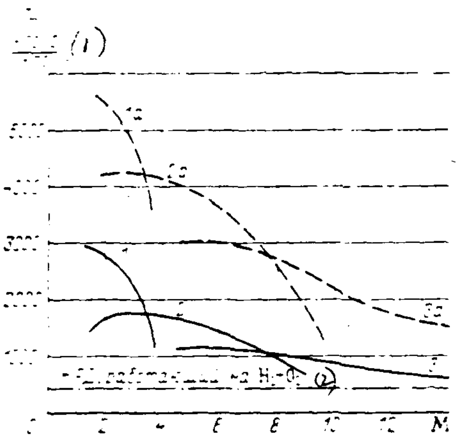

Fig. 7.6. Dependence of specific impulse (on fuel/propellant) on Mach number of flight for different types of engines, which work on kerosene (1; 2; 3) and hydrogen (1a; 2a; 3a): 1; 1a - GTD; 2; 2a - PVRD with subsonic combustion; 3; 3a - PVRD with supersonic combustion.

Key: (1). kgs·s/kgs. (2). ZhRD, which works on H_2+O_2.

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There are methods of comparison of engines according to their fundamental characteristics, for example, is examined the product of the specific parameters: $\gamma_{дв}, c_{у}, D_{дв}, P_{у}$ or is equal the total weight of engine installation and fuel/propellant $(G_{ду}+G_{топ})$ and so forth. However, these methods carry the approximate character and they do not make it possible to solve the presented question with a sufficient precision/accuracy.

In order to select one or another engine, it is necessary to make detailed airplane performance computation with each engine, after determining flight (and other) characteristics of aircraft, and to consider degree of fulfilling requirements, presented to this aircraft.

For contemporary civil/civilian and military aircraft (as a rule, heavy) specially is projected/designed engine under prescribed/assigned characteristics of aircraft. The designers of aircraft and the designers of engine carry out great joint operation on the decision of the questions, connected with installation of this engine on this aircraft.

Only after accomplishing of works indicated question of selection of engine can be solved finally.

Necessary quantity of engines for power plant of aircraft depends on whole series of factors, caused by designation/purpose of aircraft, as well as by its basic parameters and flight characteristics.

Discrepancy of effect of number of engines on safety, efficiency/cost-effectiveness and regularity of flights leads to the fact that selection of number of engines, until now, remains insufficiently worked out question of design of aircraft.

In general terms requirements for all aircraft during selection of number of engines can be formulated thus:

- aircraft must possess necessary starting thrust-weight ratio;

- aircraft must possess sufficient reliability and efficiency/cost- effectiveness.

§ 2. Air intakes of contemporary aircraft.

Functions of air intake in system of power plant of contemporary aircraft are reduced to following:

- to ensure air compression, which enters air intake, converting kinetic energy of incident flow into pressure.

As is known, at subsonic flight speeds increase in air pressure in engine channel occurs basically in compressor of TRD (approximately five times more than in diffuser). With an increase in the velocity of the function of compressor gradually they pass to the air intake; with number M=1.2-1.4 the air intake and compressor to the identical degree compress flow. At the high supersonic flight speeds (M>3) the role of compressor becomes already unessential, and compression ratio in the input device reaches order 40:1.

Compression ratio of air in turbojet engines it is accepted to call ratio of air pressure at the end of process of compression, i.e., after compressor, to atmospheric pressure

$$\pi = \frac{p_k}{p_H} = \frac{p_c}{p_H} \cdot \frac{p_k}{p_c} = \pi_{vx}\pi$$

where p_k - compressor discharge pressure (at burner inlet);

p_H - atmospheric pressure;

p_c - pressure at entry into compressor;

$\pi_{вх}$ - compression ratio in air intake;

$\pi_к$ - compression ratio in compressor.

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Change in values $\pi_{вх}$ and $\pi_к$ with increase in velocity for air intake and TRD, designed for cruising flight speed with M=3.0, flows/occurs approximately as follows:

M	0	1	2	2.2	2.5	2.7	3.0
$\pi_{вх}$	~1	2	7	10	15	20	~30
$\pi_к$	~8	6	5	4.7	4.4	4.2	~4

To air intake of supersonic aircraft at present is assigned role of adjustable compressor.

In diffuser pressure so grows (for example, when $M \approx 2.2$ $\pi_{вх} = 10$) its distribution on internal surface is such, that thrust, equal to 60-75% of entire thrust of power plant (Fig. 7.7), is created.

During braking of flow caused by friction, vortex formation (flow breakaway in nonuniform velocity field), by heat exchange always take place of loss of pressure, while during stagnation of supersonic flow appear wave losses, caused by emergence of shock waves. As a result of air-intake loss actually attainable values $\pi_{вх}$ prove to be less than theoretically possible. For example, with number M=3 it is possible to obtain $\pi_{вх} \approx 30$ instead of $\pi_{вх\,ид} \approx 38$, which would be in the ideal case (without the losses).

It is accepted to consider losses of pressure, which appear during air compression in input device, by value of recovery factor of total pressure (in theory of engines for convenience in calculations they frequently use not static, but total pressures). The total pressure recovery coefficient is equal to

$$\sigma_{в} = \frac{\pi_{в}}{\pi_{в\,ид}} = \frac{p^*_{в}}{p^*_{H}},$$

where $p^*_{в}$ - pressure of the completely stagnant flow at the end of the air intake (at the entry into the compressor);

p^*_{H} - the free-stream total head of air.

So that air intake of contemporary aircraft effectively would fulfill its functions, it must ensure:

- highest possible values of total pressure recovery coefficient;

- sufficiently uniform field of inlet velocities into compressor;

- stable (without separations of flow and pulsations of pressure) operation in all regimes of operation;

- as small as possible external resisting.

Subsonic air inlets.

The accumulated experience of construction and operation of subsonic air inlets makes it possible to obtain very high values of total pressure recovery coefficient in similar input devices — $\sigma_{вх}=0{,}97—0{,}98$.

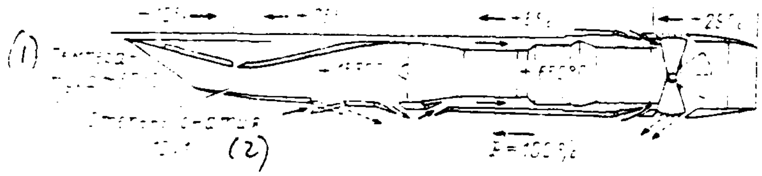

Fig. 7.7. Distribution of thrust forces and aerodynamic drag along the length of engine nacelle with number M=2.2.

Key: (1). Temperature. (2). Compression ratio.

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At design of subsonic air inlets their parameters are chosen for basic flight conditions.

Sizes/dimensions of inlet of diffuser are determined by air flow rate through intake area. According to the law of conservation of mass the weight flow rate of air per second in cross section H-H and $вх-вх$ (Fig. 7.8) will be identical:

$$G_в = F_H V \gamma_H = F_{вх} V_{вх} \gamma_{вх},$$

where V - rated speed of flight at height/altitude H.

Intake area can be expressed thus:

$$F_{вх} = \frac{G_в}{V_{вх} \gamma_{вх}}, \qquad (7.6)$$

where $G_в$ — flow rate per second by engine, under which air intake (it is prescribed/assigned in engine characteristics), is projected/designed;

$V_{вх}$ — air speed at entry into air intake;

$\gamma_{вх} = \varrho_{вх} g$ — specific weight/gravity of air at entry.

Value $V_{вх}$ in the first approximation, can be determined in the form

$$V_{вх} = V \overline{V}_{вх},$$

where $\overline{V}_{вх} = 0.3—0.7$ — relative air speed at entry into air intake.

Smaller values of value $\overline{V}_{вх}$ are accepted for long and bent channels (in order to have small hydraulic losses), larger value $\overline{V}_{вх}$ — for short channels and GTD with high inlet velocities into compressor.

Increase in air density $\rho_{вх}$ with during braking from V to $V_{вх}$ is determined according to special gas-dynamic tables.

In period of precomputations (and also during diploma design) sizes/dimensions of inlet can be determined through relative diameter of entry

$$\overline{D}_{вх}=1.1\sqrt{\frac{1}{\frac{1-\overline{V}_{вх}^2}{1-M^2}+1}}. \quad (7.7)$$

Here $\overline{D}_{вх}=D_{вх}/D_{г}$;

$D_{вх}$ — diameter of the entry of air intake;

$D_{г}$ — maximum outer diameter of nacelle, moreover

$$D_{г}\approx(1,2-1,3)D_{вн},$$

where $D_{вн}$ — maximum inner diameter of nacelle.

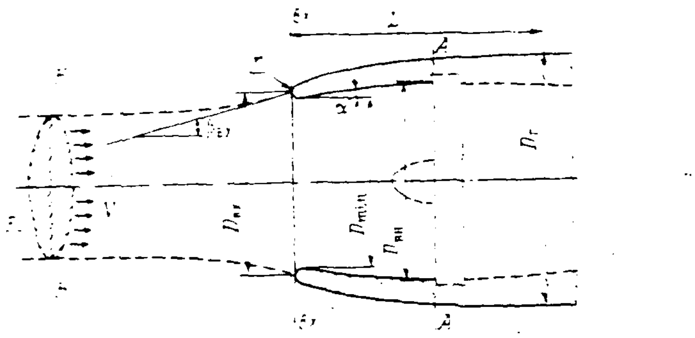

Fig. 7.8. Diagram of subsonic air inlet.

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It is possible to accept $D_{вн} \approx D_{дв}$ — the diameter of engine (on the compressor).

Shaping entering edge is accomplished/realized in order to obtai even flow of nacelle and to avoid flow separations at entry.

Angle of indraft of external jet boundary to entry into air intake relative to its axis/axle in the first approximation, can be determined through relative rate of entry

$$\beta_{вх} = 22\sqrt{1/\bar{V}_{вх} - 1}, \qquad (7.8)$$

where $\beta_{вх}$ — in degrees.

Radius of curvature of air-intake lip, which ensures even flow, approximately can be accepted

$$r_{min} = (0,04-0,05)\sqrt{F_{BX}} \qquad (7.9)$$

Shaping air duct is provided for for obtaining greatest value of value σ_{BX} and uniform portion of inlet velocities into compressor (cross section A-A).

If expansion of channel (after entry) is too great or channel has sharp rotations and bendings, flow can be removed away from walls, which will lead to considerable to eddy losses. Another reason for losses in the subsonic diffuser - air friction against the walls of channel. However, if flow breakaway does not occur, then losses from the friction prove to be comparatively small.

If diffuser is done with rectilinear walls, then half-angle of its solution/opening must be

$$\alpha \leq 4-5^{\circ}.$$

If channel has rotations and bendings, then in latter/last section (before engine) axis/axle of channel must coincide with axis/axle of compressor. The length of this cylindrical is rapid the channel it must be not less $(0.5-1)D_{BX}$.

Shaping external enclosures of air intake must ensure to it minimum drag. Therefore the external enclosures of air intake are shaped independent of internal.

Relative external length of entry, appropriate from point of view of external flow, it can be expressed as function of Mach number of flight

$$\overline{L} = L/D_r \approx 1{,}5M^2, \qquad (7.10)$$

where $\overline{L}$ - distance from nose/leading edge of nacelle to cylindrical part.

It must be noted, that the given above dependences, which determine basic parameters of air intake, are approximate. It is theoretically very complicated to take into account all special features of real flow; therefore recommendation regarding the shaping, for example air duct, they are established/installed v s basic experimentally.

Supersonic air intakes.

In supersonic air intake losses, which appear during air compression, are composed of wave losses (in system of jumps), eddy losses and losses from friction. However, the wave losses

$$\sigma_{w} = (0{,}9 - 0{,}95)\sigma_{c},$$

where $\sigma_c = \sigma_1 \sigma_2 \ldots \sigma_n = \prod_{i=1}^{n} \sigma_i$ — total pressure recovery coefficient in the system of jumps, compose basic value;

σ_i — total pressure recovery coefficient in one jump.

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Depending on form of cross section of entry supersonic air intakes can be divided into two types: two-dimensional (flat/plane) and three-dimensional (circular, semicircular, etc.). Depending on the position of oblique shocks the air intakes are of the external, internal and mixed compression (all three types they can be and two-dimensional, and three-dimensional, Fig. 7.9).

First multishock air intakes of supersonic aircraft were of internal compression. In comparison with the air intakes of internal compression they are sufficiently simple in the regulation, they do not require complex starting system, they possess weight advantages; however, efficiency in them are below (air intakes with the external compression they have also greatest aerodynamic drag). For example with M=3, is obtained

$\sigma_{вх}=0.75$ — external compression;

$\sigma_{вх}=0.95$ — internal compression;

$\sigma_{вх}=0.85$ — mixed compression.

Parameters and sizes/dimensions of supersonic air intake are chosen for basic flight conditions (as a rule supersonic cruise).

Intake area. The capacity of air intake (diffuser) is considered by the coefficient of flow rate φ, which is the ratio of the real air flow rate to maximally possible, i.e.

$$\varphi=\frac{G_{в}}{G_{в.\,max}}.$$

Coefficient of flow rate φ is numerically equal to ratio of area of air jet in undisturbed flow (cross section H-H) to intake area into air intake (*Fig.* 7.10)

$$\varphi = \frac{F_H}{F_{\text{вх}}},$$

moreover intake area of supersonic air intake is considered total cross section *вх—вх*. The area of directly entrance slit is defined as

$$F_{\text{ш}} = F_{\text{вх}} - F_{\text{т}},$$

where $F_{\text{т}}$— sectional area of body, which creates the shock envelope.

For creation of system of jumps in air intakes can be used both flat/plane bodies (wedge) - for two-dimensional inlets and circular bodies (cone, semicone, fourth of cone) - for three-dimensional air intakes of corresponding form.

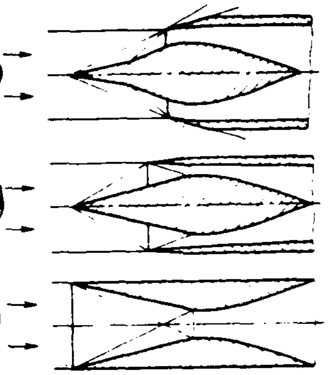

Fig. 7.9. Methods of forming shock waves: a) air intake with external compression - all oblique shocks are arranged/located outside; b) air intake of mixed compression - oblique shocks are arranged/located and outside, and within air intake; c) air intake with internal compression - all oblique shocks are arranged/located inside.

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However, geometric parameters of all air intakes are analogous; therefore longitudinal section, for example semicircular and two-dimensional inlets, it will be virtually equal, and longitudinal section of circular air intake will be characterized by only symmetry of lower and upper part of cross section.

In design conditions of work of air intake external oblique shock waves are focused on leading edge of shell, i.e. occurs equality

$$\varphi=\frac{G_{\text{в}}}{G_{\text{в}\,\max}}=\frac{F_H V \gamma_H}{F_{\text{вх}} V \gamma_H}=1,$$

$$F_{\text{вх}}=\frac{G_{\text{в}}}{V \gamma_H g}, \qquad (7.11)$$

whence

where $G_{в}$ — flow rate per second of air;

V - flight speed of aircraft;

ρ_H — air density at flight altitude;

$g=9.81$ m/s².

The complete air flow rate through the air intake is equal to

$$G_{в\Sigma}=G_{в}+G_{в.п.с}+G_{в.пер},$$

where $G_{в}$ — air flow rate through the engine;

$G_{в.п.с}$ — air of boundary layer, combined from the surfaces of compression (wedge, cone, etc.);

$G_{в.пер}$ — air, passed from the diffuser back in the atmosphere through the bypass (anti-surge) shutters/doors.

However, with a precision/accuracy sufficient for the sketch design it is possible to accept

$$G_{в\Sigma}\approx G_{в}.$$

Flow rate of air (physical) through engine is equal to

$$G_{в}=G_{в.пр}\frac{p_a^*}{p_0}\sqrt{\frac{T_0}{T_H^*}}, \tag{7.12}$$

where $G_{в.пр}$ — driven air flow rate;

$$p_a^*=p_H^*\sigma_{вх};$$

p_0, T_0 - respectively pressure and temperature of surrounding air with $H=0$; $V=0$.

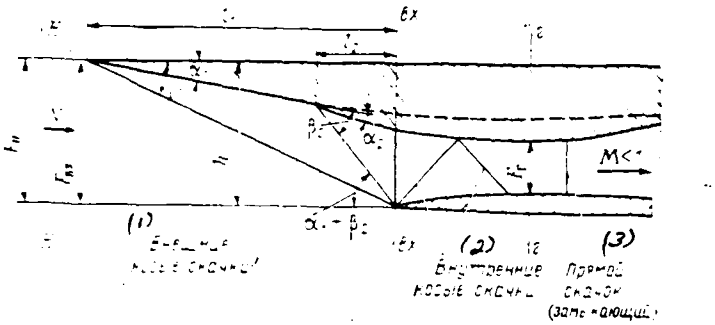

Fig. 7.10. Diagram of supersonic air intake of mixed compression (design conditions of flow).

Key: (1). External oblique shocks. (2). Internal oblique shocks. (3). Normal shock (file closer).

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Expressing pressure and total stagnation temperature through flight Mach number

$$p_H^* = p_H (1 + 0.2M^2)^{3.5}; \quad T_H^* = T_H (1 + 0,2M^2),$$

where p_H^*, T_H^* — respectively pressure and total stagnation temperature of incident flow at flight altitude;

p_H, T_H — respectively pressure (static) and temperature of surrounding air at flight altitude,

we will obtain

$$G_в = G_{в\,пр} \frac{\sigma_{вх} p_H (1 + 0,2M^2)^{3.5}}{1,033} \sqrt{\frac{288}{T_H}}. \qquad (7.13)$$

Resultant expression for determining intake area of supersonic air intake will take form

$$F_{вх} = G_{в.пр} \frac{c_{в} p_H (1 - 0,2M^2)^3}{10,14 V \rho_H} \sqrt{\frac{288}{T_H}}. \qquad (7.34)$$

where $F_{вх}$ — is expressed in m^2;

$G_{в.пр}$ — in kgf/s (value of given air flow rate they are given in appendix III);

p_H — in kg/cm^2;

V - in m/s;

ρ_H — in $kg \cdot s^2/m^4$;

T_H — in K.

Minimally necessary number of jumps in air intake of supersonic aircraft (depending on rated speed of flight) must be;

M≤1.3 - one normal shock;

M≤1.5 - system 1 oblique shock +1 normal shock;

M≤2.0 - system of 2 oblique shocks +1 normal shock;

M≤2.5 - system of 3 oblique shocks +1 normal shock;

M≤3.0 - system of 4 oblique shocks +1 normal shock;

M≤3.5 - system of 5 oblique shocks +1 normal shock.

Location of oblique shocks, as it was noted above, depends on type of air intake. During the mixed compression usually of 1-3 oblique shocks place outside, the rest - within the air intake.

Angles of stepped wedge (cone) - α_1, α_2, α_3 and so forth are chosen thus, in order to in design conditions of work of air intake external oblique shocks (the first compulsorily) shells were focused on leading edge. To focus jumps possible, is obvious, at different angles α, since the angles of the slope of jumps β depend on these angles. However, the greatest value of coefficient σ_{cr} is obtained only at the identical intensity of jumps, which is defined as the ratio of the velocity of the flow before the jump to the speed of flow after the jump.

Therefore angles α_1, α_2, α_3 and so forth must ensure equality

$$\frac{V}{V_1}=\frac{V_1}{V_2}=\frac{V_2}{V_3}=\ldots,$$

where V - speed of undisturbed flow (flight speed);

V_1 - speed of flow by 1st oblique shock;

V_2 - speed of flow after 2nd oblique shock, etc.

The speed of flow after i oblique shock V_i is connected with the speed of the flow before the abruptly following dependence:

$$V_i=V_{i-1}\frac{\cos\beta_i}{\cos(\beta_i-\alpha_i)}, \qquad (7.15)$$

where V_{i-1} — speed of the flow before the i oblique shock. For the first oblique shock, consequently, we will have

$$V_1=V\frac{\cos\beta_1}{\cos(\beta_1-\alpha_1)}, \qquad (7.16)$$

where V - flight speed.

Mach number of flow after i oblique shock is determined as follows:

$$M_i^2=\frac{5-M_{i-1}^2}{7M_{i-1}^2\sin^2\beta_i-1}+\frac{5M_{i-1}^2\cos^2\beta_i}{5-M_{i-1}^2\sin^2\beta_i}. \qquad (7.17)$$

where M_{i-1} — Mach number of flow before i oblique shock (for 1st oblique shock - Mach number of flight).

Relationship/ratio between angle of rotation of flow (by wedge angle, cone) and angle of slope of jump is expressed by formula

$$\mathrm{tg}\,\alpha_i = \mathrm{ctg}\,\beta_i \frac{M_{i-1}^2 \sin^2\beta_i - 1}{1 - M_{i-1}^2 (1{,}2 - \sin^2\beta_i)}. \quad (7.18)$$

Knowing quantity of oblique shocks, from previous equations it is possible to determine necessary values of angles of stepped wedge (cone), which ensure identical intensity in jumps.

Special attention should be paid to angle of slope of first surface of compression α_1, since it actually determines carrying out of cone (wedge) - distance from apex/vertex of wedge to intake plane.

For contemporary supersonic air intakes (depending on calculated Mach number of flight) value of angle α_1 is equal to:

M / α_1	<2.5	2.5–[illegible]
(1) Клин	~9	~[illegible]
(2) Конус	~15	~[illegible]

Key: (1). Wedge. (2). Cone.

Angle α_2 is approximately equal to angle α_1:

$$\alpha_2 = \alpha_1 \pm (0 \div 2[illegible])$$

Table 7.1. for some values of angles α gives appropriate values of angles β (depending on Mach number of flow before jump).

Length of steps/stages of cone (wedge) easily is determined, if angles α and β and size/dimension h are known.

Axial distance from intake plane to apex of the cone (wedge) will be equally

$$l_1 = h/\mathrm{tg}\,\beta_1. \tag{7.19}$$

Table 7.1.

	° \ M	1,5	2,0	2,2	2,35	2,7	3,0	3,5
(1)	5	47	34	31	29	25	23	20
	7	50	37	33	31	27	25	22
	9	54	39	35	33	29	27	24
	11	58	41	37	35	31	29	26
(2)	9	42	31	28	26	23	21	19
	11	43	32	29	27	24	22	20
	13	44	33	30	28	25	23	21
	15	45	34	31	30	27	25	23

Key: (1). Wedge. (2). Cone.

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Distance from intake plane to beginning of second step/stage of cone (in Fig. 7.10 - size/dimension l_2) let us find, after leading from point of focusing of 2nd jump ray/beam at angle $(\alpha_1+\beta_2)$ to incident flow, etc.

Throat area of air intake F_r (in cross section $r-r$) must be reduced with increase in the velocity of flight. Physically this is completely obvious; with an increase in Mach number of flight grows pressure ratio of air in the system of jumps, and consequently, are raised pressure and air density in the throat, that also leads to the need of decreasing its area (otherwise air in the throat it will be widened and σ_{BX} it will be reduced).

Necessary relative throat area

$$\bar{F}_{r}=\frac{F_{r}}{F_{BX}}.$$

Usually is calculated

Value of value F_{r} depending on Mach number of flight in the first approximation, can be taken

M	1,5	2,0	2,5	3,0	3,5
$\bar{F}_{r}$	0.5	0,42	0,35	0,32	0,3

Parameters of supersonic air intake obtained as a result of sketch design compulsorily are checked and are corrected in process of experimental tests.

Regulation of supersonic air intakes.

Supersonic air intakes must ensure high values of total pressure recovery coefficient σ_{BX} in considerably larger speed range, than subsonic air intakes. Therefore they have a control system, whose task consists of ensuring of the matched operation of air intake and engine (the capacity of air intake it must correspond to the capacity of engine).

Otherwise unstable fluctuating work (surge, "itch/buzzing" of air intake) can arise. In this case coefficient σ_{BX} strongly is reduced At supersonic speeds the task of air inlet control is reduced to hold down/retain the system of jumps (especially terminal normal shock after the throat) in the prescribed/assigned position. This can be made a change in the throat area and a bypass of excess air into the surrounding atmosphere. Air bleeding into the atmpsphere is accomplished/realized by a opening the special shutters/doors, established/installed on the canal surface (after the throat) of air intake. These shutters/doors were called anti-surge or bypass. During the supersonic cruise the shutters/doors of air bleeding are opened slightly and the part of air is removed/released into the atmosphere, preventing thereby the emergence of the surge of air intake.

During takeoff and at small subsonic flight speeds required throat area proves to be more than value $F_{\text{r max}}$. determined by structural/design possibilities (this it is explained by relatively low air density in throat). Therefore in spite of the completely revealed throat, to engine for the normal operation air is not sufficient. In order on to break the regime of the work of engine, during the takeoff and at the low subsonic flight speeds auxiliary (takeoff) shutters/doors are opened/disclosed additionally and supplementary air proceeds to engine, omitting the throat (auxiliary and bypass shutters/doors they are shown in Fig. 7.16).

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Structurally/constructively control system is implemented:

a) for changing the throat area:

- by displacement/movement of cone forward - back/ago (circular air intakes);

- by displacement/movement of mobile manifolds/ramps (the two-dimensional inlets);

- by change in the diameter of inner body (circular air intakes);

b) for the further suction or the air bleeding:

- by opening of further openings/apertures in the channel after the throat (auxiliary and bypass shutters/doors - on all supersonic air intakes).

Air inlet control is accomplished/realized by automatic system. The schematic of a similar system is shown in Fig. 7.11. In the system there are two main chains of control of inner body and shutters/doors of bypass, with the help of which is accomplished/realized the regulation of Mach number in the throat and the positions of the closing shock wave.

3. Arrangement/position of air intakes on the aircraft.

On contemporary aircraft engines frequently are placed in special external nacelles, where air intake directly borders on compressor of engine. In this case the layout of engine and air intake is implemented together with the layout of engine nacelle.

During engine installation within fuselage or wing air intake is separated/liberated from engine by air duct and layout of air intake is implemented separately.

By main requirement, presented to layout of air intakes on jet aircraft, as it was already noted, it is: guarantee of uniform field of inlet velocities into compressor and obtaining high values of total pressure recovery coefficient

Essential nonuniformity of field of velocities of flow can cause vibration of compressor blades and their breakage. Even the permissible nonuniformity of the field of velocities decreases the service life of compressor and engine as a whole. Friction (caused by the viscosity of air) is the basic source, which calls the nonuniformity of the field of the velocities in the air intake. The presence of friction, as is known, causes the appearance of the boundary layer on the fairing, the velocity in which sharply falls from the velocity of the undisturbed flow to zero.

During supersonic flow boundary layer, interacting with shock waves, breaks their clearness; appear local separations of flow from walls; boundary layer, passing through jumps, even more greatly increases its thickness; in places of bulging boundary layer new weak oblique shocks (λ-jumps), etc are formed.

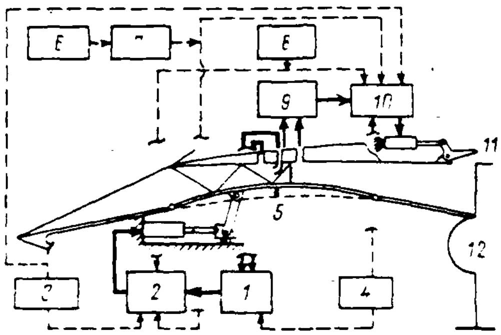

Fig. 7.11. Hypothetical control system of supersonic air intake (they are shown basic and auxiliary functions): 1 - sensor of Mach number in throat; 2 - regulator of inner body; 3 - manual control; 4 - sensor of surge; 5 - inner body; 6 - sensor of "disruption/separation" ' 7 - launch control; 8 - sensor of Mach number; 9 - position detector of jump; 10 - regulator of bypass shutters/doors; 11 - bypass shutters/doors; 12 - engine: - - - → - basic functions; - - - - - → - auxiliary functions.

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Deviation from design diagram of flow, caused by viscosity of air, in the final analysis leads to nonuniformity of field of velocities and reduction/descent σ_{rx} Therefore all contemporary air intakes have a system of the branch/removal (drain) of boundary layer. Recedes both the boundary layer, which was being formed not of the surface of fuselage (or wing) and the boundary layer, which arose on the surfaces of compression - cone (wedge) and the internal surface of shell (Fig. 7.12).

Boundary layer thickness δ, as is known, depends on velocity of flow, on coefficient of viscosity of air and on length of contact of flow with washed surface

$$\delta = f(V, \nu, x)$$

During design of air intake for reliable removal/distance of boundary layer they accept the height of drain slots (h_1, h_2...)

$$h \approx 0.01 l.$$

where l - length of surface, at which is formed boundary layer.

If, for example, air intake will close adjoin to surface of fuselage (i.e. h_1=0), then total pressure recovery coefficient with M=2.5 decreases by 25-30%, which will bring in the final analysis to reduction in engine thrust to ~45% and to increase in specific fuel consumption per ~15%.

For final equalization of velocity fields after throat of supersonic air intake vortex generators (small plates) are established/installed. Place and need for the installation of vortex generators is determined in the process of the finishing of air intake (during the wind tunnel tests).

Guarantee of boundary-layer bleed is one of special features of layout of contemporary air intakes.

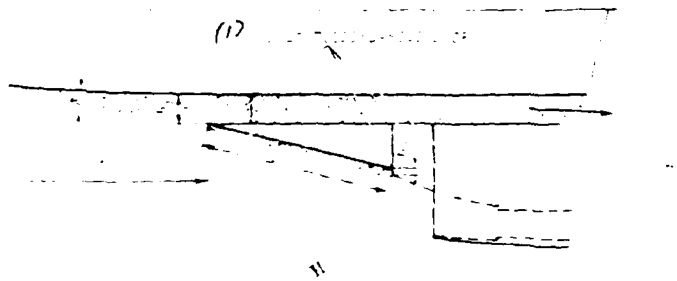

Fig. 7.12. Diagram of branch/removal of boundary layer.

Key: (1). Boundary-layer bleed.

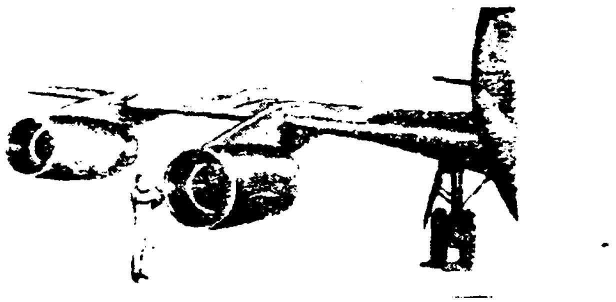

Fig. 7.13. Pylon suspension TRDD under wing.

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Depending on place of arrangement/position on aircraft are used following basic types of air intakes:

1) formtal air intakes (mainly, circular);

2) off-axis inlets (circular, semicircular, flat/plane, etc.);

3) wingtip air intakes (mainly, flat/plane).

Frontal air intakes are placed either in to nose of fuselage on light aircraft or in to nose of engine nacelle, suspended/hung from pylon under wing of heavy aircraft (Fig. 7.13). The major advantage of frontal air intakes consists in the fact that they ensure the high uniformity of velocity fields, and during the supersonic flight in the design conditions, furthermore, they make it possible to strictly maintain/withstand the prescribed/assigned position of the shock envelope.

However, frontal air intakes have number of shortcomings. If on the heavy nonmaneuverable aircraft during entire cruise angle of attack does not vary, and consequently, the system of jumps at the entry into the air intake retains the prescribed/assigned position, then on the light aircraft with maneuver accomplishment with the heavy overload, when angle of attack considerably increases, the focusing of jumps is broken, which leads to the nonuniformity of the field of velocities and to reduction in the total pressure recovery coefficient. Briefly stated, frontal air intakes not high angles of attack work insufficiently effectively.

Second shortcoming in frontal air intakes carries layout character. Placing air intake in to the nose of fuselage, it is necessary to occupy large internal volumes in the fuselage under the air duct (actual entire fuselage from the nose to the tail it is cut by air and engine channel), which is natural, it complicates the layout of aircraft. Furthermore, frontal air intake does not make it possible to place in to the nose of fuselage the antenna of large-diameter radar (antenna it is limited to the sizes/dimensions of intake cone).

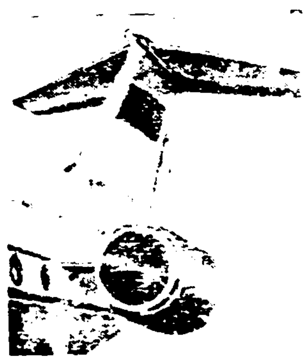

Fig. 7.14.

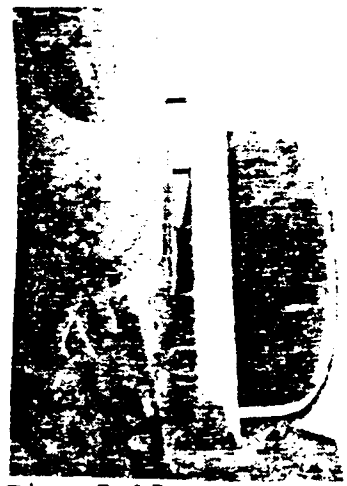

Fig. 7.15.

Fig. 7.14. Engine nacelle of aircraft of Tu-134.

Fig. 7.15. Air intake of fighter-bomber Phantom-II (on surface of fuselage is visible bay for positioning/arranging rocket "Sparrow". On inside of shell is established/installed PVD of the control system of air intake).

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Off-axis inlets in entry form differ in terms of great variety. On the subsonic aircraft in essence are used either the circular air intakes or the air intakes, the entry form of which is close to the rectangular (deviation from the circle in this case it is explained by the tendency to preserve wing profile); these air intakes are called also wing, since they are actually arranged/located in root of the wing.

Presence of boundary layer on surface of fuselage requires creation of drain slots with layout of off-axis inlets on aircraft, for which entire engine nacelle is moved aside and is fastened to fuselage to pylon (Fig. 7.14).

It must be noted that pylon in this case makes it possible to only organize drain of boundary layer, without ensuring, however, working conditions of frontal air intake, since slope of flow after wing, wake from wing and other aircraft components create local conditions for flow, different from undisturbed flow.

On supersonic aircraft in essence are used flat/plane (Fig. 7.15) and semicircular off-axis inlets (although they are encountered and other entry forms. The arrangement/position of air intakes along the sides of the fuselage not only considerably shortened the length of air duct, but also it completely freed the nose of fuselage for the installation of radar. During the precise organization of boundary-layer bleed the off-axis inlets work very effectively (however with maneuver accomplishment at supersonic speed with the large slip angles one of the air intakes it can prove to be the darkened forward fuselage).

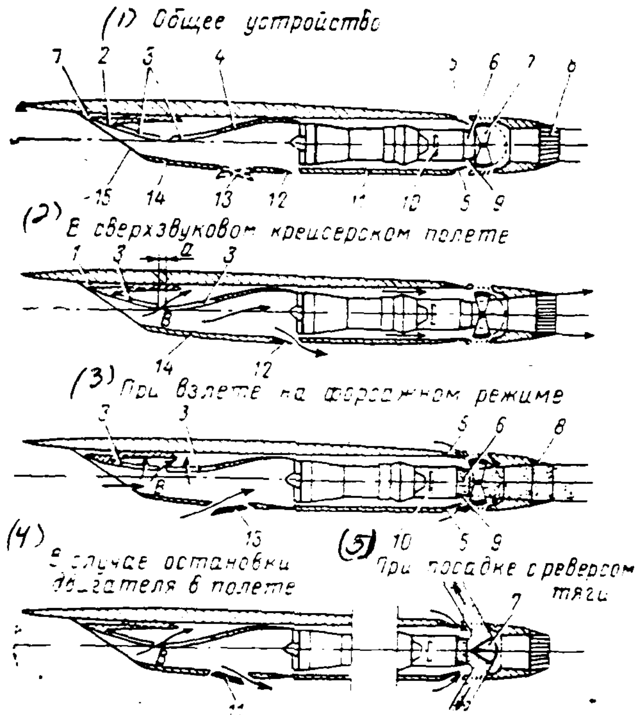

Fig. 7.16. General-arrangement diagram of engine nacelle of heavy supersonic aircraft: 1 - slot for boundary-layer bleed from wing surface; 2 - fixed plane of wedge; 3 - mobile manifolds/ramps; 4 - subsonic air intake duct; 5 - auxiliary air intake; 6 - primary nozzle; 7 - reverser of thrust; 8 - adjustable secondary nozzle; 9 - silencer; 10 - injector of afterburner; 11 - wall under engine; 12 - bypass (anti-surge) shutter/door; 13 - auxiliary (takeoff) shutter/door; 14 - shell; 15 - vertical partition/baffle, which divides air intakes of two engines; a) slot for boundary-layer bleed from plane of wedge; c) air of boundary layer.

Key: (1). General arrangement. (2). In supersonic cruising flight. (3). During takeoff in afterburning regime. (4). In the case of engine shutdown in flight. (5). during thrust reversal landing.

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Wingtip air intakes on existing aircraft (KhV-70, Tu-144, "Concorde") have flat/plane entry. Air intake in this case is the forward section of the engine nacelle, established/installed under the wing. Fig. 7.16 shows the longitudinal section of the engine nacelle of aircraft "Concorde".

Shortcoming in wingtip air intakes is poor work on large negative angles of attack (in this case they they are shaded by wing).

§ 4. The fuel system of aircraft.

Fuel system of contemporary aircraft includes in itself following basic elements: fuel tanks, conduits/manifolds, pumps, valves, taps/cranes, filters and system of different automatic machines, sensors, measuring meters, etc.

Designation/purpose of fuel system - to ensure supply of fuel/propellant to engines in all possible for this aircraft flight conditions (on height/altitude, velocity, g-forces, etc.) in necessary quantity and with necessary pressure. On the supersonic aircraft (especially intended for the prolonged supersonic flight) the fuel system implements another series/row of important functions, providing cooling conditioning system, hydraulic system, etc., and also it can reduce the margin of the stability of aircraft, parrying the shift/shear of focus upon transfer from the subsonic to the supersonic flight (see Chapter VI).

As main fuel for jet engines of contemporary aircraft is used it is used hydrocarbon fuel (improved types of kerosene). For the hypersonic and aerospace aircraft cryogenic fuels/propellants (in essence liquid hydrogen) are of interest.

At high flight velocities, when direction of heat flux in construction of aircraft is changed to opposite, fuel/propellant (especially placed in wing tank compartments) is heated. For example, in the cruise at a velocity, which corresponds to number M=3, the heat flux, which proceeds from the skin/sheathing, becomes this essential (mean temperature of skin/sheathing ~260°C) that it could heat kerosene to the boiling point. The application of the thermal insulation (laminar honeycomb sandwich construction also considerably decreases heat flux) makes it possible to avoid the excessive heating of fuel/propellant; however, its temperature at high velocities (M>3) nevertheless so is raised that hydrocarbon fuel cannot perform the role of coolant for cooling the construction/design and the system of aircraft.

One of the shortcomings in hydrocarbon fuel is its thermal instability at high temperatures (for example, for kerosene this limit it is equal to approximately 200°C). if the limit indicated is reached even to the short period, then can be formed a certain number of deposits of solid particles in the heat exchangers, the filters, the injectors of engines and so forth with their possible blocking. In this connection one should speak about the superiority of liquid hydrogen as fuels/propellants for the hypersonic aircraft.

At hypersonic speeds hydrogen before supply into engine can be used for cooling construction of aircraft. And then considerable difficulties, connected with the application of hydrogen, will be to some degree compensated. Matter in the fact that liquid hydrogen, possessing the irrefutable advantage over the kerosene: energy content per unit weight is approximately 2.5 times more, specific heat is 7 times higher (liquid hydrogen - distinct coolant), has the number of essential shortcomings.

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A larger shortcoming in hydrogen - low density in the liquid state, which constitutes 0.1 of density of kerosene; therefore energy content per unit volume in hydrogen is approximately 4 times less. Another shortcoming is the low boiling point of liquid hydrogen, equal to -253°C. Therefore flight vehicles with this fuel/propellant will have large volumes, moreover fuel/propellant must be placed in the cryogenic tanks, which additionally increase weight and volume of apparatus.

Comparison of characteristics of kerosene and hydrogen is given in Table 7.2.

Differences in physical and thermodynamic properties of hydrogen and kerosene have great effect on possible compromise solutions, which link flight characteristics and construction of aircraft.

Arrangement of fuel to a considerable extent determines overall design of aircraft, since fuel reserve not contemporary aircraft can reach 50% and more from takeoff weight (on Il-62 value $\bar{G}_{T.max}=0.51$; and XB-70 $-\bar{G}_{T.max}=0.57$) is placed fuel/propellant in the special fuel tanks, which are divided into the bases, the expenditure and the balancing (depending on designation/purpose and aircraft type expenditure and balancing tanks in the fuel system they can and not be provided for).

According to design criteria fuel tanks are divided into three types; rigid, soft and tank compartments of construction of aircraft. The major advantage of soft rubber tanks consists in the fact that they allow rigid tanks, to better than use a volume, are more technologically effective and convenient in the production and the operation (it is possible to wrap up them and to install through handholes). Flexible tanks, furthermore, they do not fear vibrations, do not give torn edges with the leakage (small holes even they are involved/tightened by a special layer of rubber), possess good thermal insulation properties.

Smaller weight in comparison with soft tanks (together with containers) is advantage of rigid tanks. Rigid tanks are autonomous from the construction of the aircraft, it is possible to comparative-- easily repair them.

Hermetically sealed tank compartments (tank-construction/design) make it possible to most rationally use internal volumes of aircraft, since there is no fuel tank as such, and fuel/propellant fills into section of wing or fuselage, covered with from within kerosene resisant (and temperaturechange resisting) sealing compound. The application of tank compartments makes it possible to increase the fuel reserve on board the aircraft. However, tank compartments possess an increased vulnerability, which depresses the reliability of fuel system, it is difficult to repair them. Fuel/propellant in the tank compartments directly undergoes the effect of low temperatures i- flight at subsonic speeds and of high temperatures in flight at the high supersonic velocities.

Table 7.2.

(1) Характеристика топлива	(2) Керосин	(3) Водород
(4) Теплота сгорания в ккал/кгс	10 290	28 700
(5) Теплота сгорания в ккал/л	8 700	2 100
(6) Плотность в жидком состоянии в г/см³	0,833	0,0735
(7) Удельная теплоемкость в ккал/кгс·град	0,46	2,7—3,7
(8) Температура кипения (при давлении 1 кгс/см²) в °C	+187—+237	—253

Key: (1). Propellant property. (2). Kerosene. (3). Hydrogen. (4). Heat of combustion in kcal/kgf. (5). Heat of combustion in kcal/l. (6). Density in the liquid state in G/cm³. (7). Specific heat, in kcal/kgf·deg. (8). boiling point (at pressure 1 kg/cm²) in °C.

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One of basic requirements for layout of fuel tanks on aircraft is guarantee of center-of-gravity location of aircraft when manufacturing fuel/propellant within permissible limits. If aircraft is intended for the prolonged supersonic flight, then in the fuel system of this aircraft for decreasing reserve of stability can be provided the installation of the special balancing tanks (tanks, arranged/located on the largest possible removal/distance to in front and from behind from the center of gravity of aircraft); by the pumping over of fuel/propellant from the front/leading balancing tanks into the rear it is possible to approach the center of gravity the focus of the aircraft (it stored up stability it is possible to decrease also by the forced shift/shear of focus forward). If aircraft short-term leaves to supersonic speed, then the installation of the system of the decrease of the reserve of directional stability of the pumping of fuel/propellant on this aircraft, obviously, it is unsuitable.

Fig. 7.17 shows typical change in center-of-gravity location and focus of aircraft in dependence on velocity (or time) of flight.

Permissible range of centering is retained in flight because of symmetrical location of fuel tanks relative to center of gravity of aircraft and specific sequence of consumption of fuel/propellant of them. Can arise the question: it is not possible whether to decrease value to ensure this order of the consumption of fuel/propellant so that the center of gravity at supersonic speed would pass position ($\bar{x}_{T}$) established/installed maximally rear with M<1 and it did approach the focus of aircraft, for example due to the consumption of fuel/propellant only from the front/leading tanks? However, this it is not possible to do for two reasons: first, if fuel/propellant from the front/leading tanks will be produced (but not to be pumped over into the rear), then the center of gravity it will be shifted/sheared back/ago very slowly and effect from the decrease of reserve of stability considerably it will be reduced; secondly, when the center of gravity it will be located from behind position $\bar{x}_{T}$ and for any reason it will be necessary to rapidly reduce the velocity and to pass on the subsonic to the subsonic position (in the opposite case aircraft it will become unstable), but to rapidly return the center of gravity to the subsonic position it is possible only by the pumping of fuel/propellant into the front/leading tanks. Therefore the system of the shift of the center of gravity of aircraft with the help of the pumping over of fuel/propellant from the front/leading balancing tanks into the rear compulsorily must guarantee and the rapid reverse/inverse pumping of fuel/propellant upon transfer from the supersonic flight to the subsonic. On the aircraft "Concorde" during the dispersal/acceleration in the transonic regime in 5 min it is approximately pumped over by 9200 l of fuel/propellant of four front/leading balancing tanks into the rear tank. The reverse/inverse pumping of fuel/propellant upon transfer to the subsonic flight speed is produced still more rapid - in 4 min.

Feed/supply of engines by fuel/propellant can be accomplished/realized from any fuel tanks, but most frequently for this are used special service tanks, into which in determined order is pumped over fuel/propellant from basic tanks.

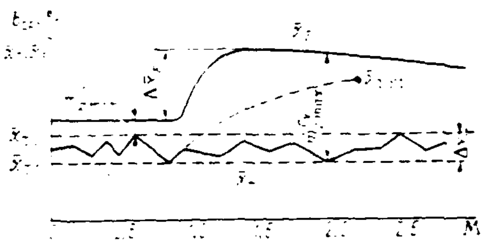

Fig. 7.17. Typical change in center-of-gravity location and aerodynamic focus of aircraft in process of flight: $\bar{x}_{T}$- maximally front/leading position center of gravity; $\bar{x}_{T}$- maximally rear position center of gravity; $\Delta\bar{x}_{T}$- permissible range of position of center of gravity in aircraft; $\Delta\bar{x}_{F}$- shift of aerodynamic focus upon transfer from subsonic to supersonic flight; $\bar{x}_{T}$- possible position center of gravity in supersonic cruise (after pumping of fuel/propellant from front/leading balancing tanks in the rear).

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Fuel system with service tank has definite advantages, especially for aircraft of military designation/purpose; it possesses larger reliability, since upon entry from system of basic tanks in reserve remains service tank (to 20% of entire fuel/propellant), whose protection it is possible to provide via armoring; one service tank (or to somewhat more easily equip with special devices/equipment for uninterrupted feed/supply of engine with accomplishing of acrobatic maneuvers, etc.

Fuel tanks can be placed both in fuselage and in wing of aircraft. Both that and other layout has definite advantages and shortcomings. Therefore for some aircraft fuel/propellant is placed in essence in the wing, for others - in the fuselage. Wing fuel tanks have the large area of the beaten surface, which leads to the smaller life of fuel system. This is main disadvantage in this location of fuel tanks. However, during the arrangement of fuel in the wing its weight unloads wing in flight, thanks to which the specific gain in the weight of wing construction is obtained. Furthermore, during the arrangement of fuel in the wing fuselage virtually completely can be occupied under the payload which has high value first of all for the passenger and transport aircraft. Therefore on all passenger, transport and heavy military aircraft fuel tanks are placed mainly in the wing (Fig. 7.18).

Arrangement of fuel in fuselage can prove to be more appropriate for military aircraft (especially flying low). Positioning fuel tanks in the fuselage above the center section, the engine, the chassis/landing gear, etc., which serve seemingly protective shield from the antiaircraft fire, it is possible to raise the life of aircraft.

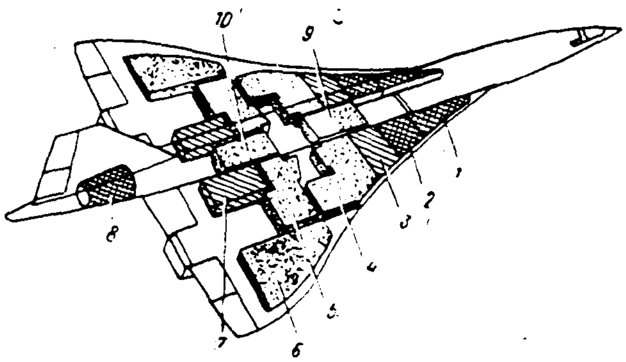

Fig. 7.18. Arrangement/position of fuel tanks on supersonic passenger aircraft "Concorde", France- England; 1; 2; 8 - balancing tanks; 3; 7 - service tanks; 4, 5, 6, 9, 10 - basic tanks.

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CHAPTER VIII.

DETERMINATION OF THE BASIC PARAMETERS OF AIRCRAFT.

In present chapter determination of basic parameters mainly of military aircraft is given. The determination of the parameters of passenger aircraft, vertical-taking off and air space aircraft is given in the special chapters of the book.

As already mentioned, they are basic parameters of aircraft:

- complete (takeoff) weight G_0;
- wing area S;
- thrust P_0 or power N_0 SU, required for obtaining prescribed/assigned flight-performance data.

For determining these parameters on initial stage of design of aircraft relative parameters p_0 and $\bar{P}_0$ are found.

§ 1. Determination of the value of the specific wing load.

In Chapter II it was shown that value of specific wing load p_0 substantially affects its takeoff weight G_0 aircraft performance and. In particular, the prescribed/assigned takeoff and landing characteristics can be decisive during determination of p_0. The takeoff and landing characteristics of aircraft include:

the landing speed

$$V_{\text{пос}} = A\sqrt{\frac{p_{\text{пос}}}{c_{y\,\text{пос}}}};$$

unstick speed during the takeoff

$$V_{\text{отр}} = B\sqrt{\frac{p_{0}}{c_{y\,\text{отр}}}};$$

approach speed

$$V_{\text{зах}} = C\sqrt{\frac{p_{\text{пос}}}{c_{y\,\text{зах}}}},$$

where A, B, C - the constant coefficients, which consider air density and ground effect on aircraft;

$p_{\text{пос}}$ — the specific wing load during the landing;

$c_{y\,\text{пос}}$; $c_{y\,\text{отр}}$ — lift coefficients which correspond to the landing angle of attack and to angle of attack with the breakaway of aircraft;

$c_{y\,\text{зах}}$ — lift coefficient at the angle of attack, which corresponds to approach speed.

High velocity $V_{\text{пос}}$ leads to need for creation of large airfields and complicates aircraft handling during landing. High velocity leads to an increase in the sizes/dimensions of airfield or to the need of applying the takeoff accelerators, while high velocity $V_{\text{отр}}$ impedes piloting during landing, which decreases the operating $V_{\text{зах}}$ characteristics of the projected/designed aircraft.

Permissible value $V_{\text{пос}}$ depends on designation/purpose of aircraft and possibility of applying on it of landing parachutes and braking devices similar to them, and also reverse-thrust devices.

In the first approximation, on basis of statistical data, published abroad, mono to take following values $V_{пос}$[1]:

for military aircraft ... of 180-250 km/h

for military transport aircraft ... 120-150 "991n1

for trainer and sports airplanes ... 60-100 ".

FOOTNOTE [1]. Maximum value $V_{пос}$ is limited to the permissible for the chambers/cameras of wheels peripheral speed ($V_{пос}$ less than 400 km/h). ENDFOOTNOTE.

The permissible velocity $V_{зах}$ usually is determined from conditions of flight safety. For the military aircraft it is possible to take the following values:

$$V_{зах}=200-280 \text{ km/h}.$$

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It is usually considered that velocity $V_{зах}$ must exceed the velocity of disruption/separation, which corresponds to $c_{y\,max}$, to 20-30%, i.e.

$$c_{y\,зах}=\frac{c_{y\,max}}{(1,2\div 1,3)^2}=(0,7\div 0,6)\,c_{y\,max}.$$

In the first approximation, value p_0 should be chosen in accordance with the velocity $V_{пос}$ or $V_{зах}$.

Since load (fuel/propellant, jettisonable loads, etc.) expendable in flight composes 25-60% of takeoff weight and, consequently, aircraft up to moment/torque of landing proves to be considerably lightened, then is expedient value of specific load $p_{пос}$, determined from weight of aircraft at moment of landing, to connect with $V_{пос}$ and $V_{зах}$. Taking into account this, it is possible to use for determining the value p_0 the following formulas:

$$p_0 = \frac{c_{y\,пос} V_{пос}^2}{180\,(1 - 0{,}8\overline{G}_т - \overline{G}_{с.г})}, \qquad (8.1)$$

or

$$p_0 = \frac{c_{y\,зах} V_{зах}^2}{208\,(1 - 0{,}8\overline{G}_т - \overline{G}_{с.г})}, \qquad (8.2)$$

where $\overline{G}_т$ – the over-all payload ratio of fuel/propellant $(\overline{G}_т = G_т/G_0)$;

$\overline{G}_{с.г}$ – the over-all payload ratio of jettisonable loads $(\overline{G}_{с.г} = G_{с.г}/G_0)$.

In the absence on initial stage of design of blowoff data it is possible to use following approximate values $c_{y\,пос}$:

- for straight wings with powerful mechanism (slat and double-slotted extension flap) $c_{y\,пос} = 2.2–2.5$;

- for sweptback wings (χ=25-35°) with powerful mechanism device $c_{y\,пос} = 1.8–2.0$;

- for sweptback wings (χ=40-45°) with movable double-slotted flaps and slats $c_{y\,пос} = 1.5–1.8$;

- for delta wings ($\chi_{п.к} = 55–60°$) with powerful mechanism $c_{y\,пос} = 1.0–1.2$.

Necessary value of specific load p_0 in majority of cases is determined from landing conditions. At the same time one should check, how obtained a value p_0 provides other prescribed/assigned flight characteristics. If, for example, is prescribed/assigned cruising speed or number $M_{\text{крейс}}$ at height/altitude $H_{\text{крейс}}$ ($\Delta_{\text{крейс}}$), then

$$p_0 = 3960 M^2_{\text{крейс}} \Delta_{\text{крейс}} \sqrt{\frac{c_{x_0}}{D_0}} \quad \text{kgf/m}^2, \tag{8.3'}$$

where

$$c_{x_0} = c_{x_p} k_0 + c_{x\,\Phi} \frac{p_0}{k_1}; \tag{8.3}$$

$$k_0 = 1.35; \; k_1 = \frac{G_0}{\Sigma S_{\text{м}}}.$$

Taking into account (8.3') formula (8.3) is reduced to a quadratic equation relative to p_0. During the solution real root ($+p_0$) is taken.

Are given below exemplary/approximate statistical values of specific loads p_0 on wing:

fighters of normal diagram ... 400-600 kgs/m²;

fighters of tailless diagram ... 250-300 "

bombers average/mean ... 350-550 "

bombers heavy ... 550-650 "

military transport aircraft with TRD heavy ... 500-650 "

lungs transport, sport and trainer aircraft ... 150-180 "

training, transfer (with TRD) ... 100-150 "

aircraft for agriculture ... 80-120 ".

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§ 2. Determining the required thrust-weight ratio of aircraft.

During design required thrust-weight ratio of aircraft $\overline{P}_0 = P_0/G_0$ is determined usually from guarantee of prescribed/assigned conditions:

- velocities (Mach number) of flight at rated altitude $H_{расч}$;
- takeoff run lengths or accelerate-stop distance;

to number M<1 to velocity, which corresponds to number $M_{расч}>1$, and also from other conditions.

As calculated thrust-weight ratio value $\overline{P}_0$ greatest of these conditions is accepted.

With assigned magnitudes $M_{расч}$, $H>11$ km and taken off value p_0 required thrust-weight ratio

$$\overline{P}_0 = \frac{4650 M^2_{расч} c_x}{\xi p_0}, \tag{8.4}$$

where

$$c_x = c_{x_0} - D_0 \frac{p^2}{294 \cdot 10 M^4_{расч} \Delta^2}; \tag{8.5}$$

$$\xi = \frac{P}{P_0} \text{ (см. стр. 28)};$$

Δ - relative density of air at base altitude H.

If is known cruising number $M_{\text{крейс}}$ at altitude $H_{\text{крейс}}>11$ km, then starting required thrust-weight ratio

$$\overline{P}_0=\frac{6950 M^2_{\text{крейс}} c_{x_0}}{\xi p_0}, \qquad (8.6)$$

and value $H_{\text{крейс}}$ can be found on $\Delta_{\text{крейс}}$:

$$\Delta_{\text{крейс}}=\frac{1{,}76\sqrt{D_0 c_x}}{\xi \overline{P}_0}. \qquad (8.7)$$

for aircraft, intended for low altitudes, starting thrust-weight ratio is determined from following formula:

$$\overline{P}_0=\frac{7200 M^2 c_x}{\xi p_0}, \qquad (8.8)$$

where $c_x=c_{x_0}+D_0\frac{p_0^2}{525\cdot 10^5 M^4}$,

(Mach number prescribed/assigned).

From condition of guaranteeing prescribed/assigned takeoff run length starting thrust-weight ratio is equal to

$$\overline{P}_0=\frac{p_0}{0{,}9 c_{y\,\text{отр}}\rho_0 g L_{\text{разб}}}+1{,}1 f_{\text{кач}}-0{,}033, \qquad (8.9)$$

where $c_{y\,\text{отр}}=1{,}1-1{,}2$ — for supersonic aircraft of normal diagram with wing of moderate sweepback (35-45°);

$c_{y\,\text{отр}}=0{,}65-0{,}67$ — for aircraft of diagram "bobtailed aircraft" with delta wing;

$c_{y\,\text{отр}}=1{,}8-2$ — for military transport subsonic aircraft, and also light multipurpose aircraft;

$f_{\text{кач}}$ — rolling friction coefficient of wheels (during takeoff with concrete runway $f_{\text{кач}}=0.03$; during takeoff from unpaved airfield $f_{\text{кач}}=0.1$).

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from safety condition of takeoff (after breakaway) with one failed engine ($n_{\text{дв}}\geqslant 2$)

$$\overline{P}_0=\frac{1.5}{1-\frac{1}{n_{\text{дв}}}}\left[\left(\frac{c_x}{c_y}\right)_{\text{отр}}+\sin\theta\right]. \tag{8.10}$$

Here θ - smallest permissible climb angle

$$\sin\theta=0.024 - \text{для}^{(1)}\ n_{\text{дв}}=2;\quad \sin\theta=0.027-\text{для}\ n_{\text{дв}}=3;$$
$$\sin\theta=0.03-\text{для}\ n_{\text{дв}}=4.$$

KEY: (1). for.

For majority of types of supersonic aircraft (except VTOL aircraft and STOL) starting thrust-weight ratio can be determined according to following formula:

$$\overline{P}_0=\frac{1.2}{K_{\text{крейс}}}\left[1+\sqrt{0.3+\frac{(0.95-\overline{G}_{\text{т.расх}})(H_{\text{крейс}}+V_{\text{крейс}}^2/2g)K_{\text{крейс}}c_{p.\text{крейс}}}{\varepsilon V_{\text{крейс}}\gamma_{\text{дв}}\overline{G}_{\text{т.расх}}}}\right], \tag{8.11}$$

where $K_{\text{крейс}}$ — aerodynamic aircraft quality/fineness ratio at cruising speed;

$\overline{G}_{\text{т.расх}}=G_{\text{т.расх}}/G_0$ — over-all payload ratio of fuel/propellant, spent for time of entire flight;

$H_{\text{крейс}}$ — initial cruising height m;

$V_{\text{крейс}}$ — cruising speed in m/s;

$c_{p.\text{крейс}}$ — specific hourly consumption of fuel/propellant in cruise in kg/kg·h;

$\gamma_{\text{дв}}$ — starting weight per horsepower.

Values of dimensionless coefficient ϵ and value $K_{крейс}$ depending on cruising Mach number are taken from following Table [1]);

$M_{крейс}$	2—3,5	6,0	9,0
ϵ	2700	3500	4700
$K_{крейс}$	6—91	5—7	4—6

FOOTNOTE [1]. Smaller values $K_{крейс}$ are characteristic for the aircraft of short range, larger values - for the aircraft of long range. ENDFOOTNOTE.

FOOTNOTE [1]. V. F. Mishin. Selection of engine in the period of the preliminary development of aircraft. Publ. the MAI, 1968. ENDFOOTNOTE.

§ 3. Determination of the complete (takeoff) weight of aircraft.

Determination of takeoff weight of aircraft - one of basic tasks during the first stage of design. The degree of accuracy, with which is determined G_0 [2], has special importance, since this value affects basic flight data of aircraft.

FOOTNOTE [2]. *The approximate determination G_0 here is intended.* ENDFOOTNOTE.

First of all it is necessary to keep in mind that wing area S is determined on the basis of value of specific load p_0 selected for designing aircraft; therefore examining effect on flight characteristics of gross weight G_0, it is necessary to accept always p_0=const.

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Overestimate of takeoff weight of projected/designed or constructed aircraft always worsens/impairs its flight characteristics. Calculation is shown, for example, that with a gain in weight G_0 by 10% number M_{max} is reduced at supersonic speeds by approximately 12%. The overestimate of weight G_0 can be obtained or in the initial stage of the design of construction/design. If the overestimate of weight was obtained only as a result of the weight increase of construction/design, then with invariability G_0 $\bar{G}_K$ it will be increased, $\bar{G}_T$, $\bar{G}_{cy}$ and thrust-weight ratio $\bar{P}_0$ to be reduced and flight characteristics will deteriorate.

As is known, gross weight of aircraft includes empty weight $G_{пуст}$ and weight of full load G_H. In turn empty weight $G_{пуст}$ includes the weight of the structure of aircraft G_K, the weight of power plant G_{cy}, the weight of equipment and control $G_{об.упр}$. The weight of full load G_H consists of the fuel load G_T, payload weight $G_{пн}$, by represented by itself the weight of different kind of loads, passengers and weight of official load $G_{служ}$ (crew and equipment) (Fig. 8.1). Thus,

$$G_0 = G_{пуст} + G_H = G_K + G_{cy} + G_{об.упр} + G_T + G_{пн} + G_{служ}.$$

Determination of gross weight is complicated by fact that some of its components are functions of weight itself G_0, and besides that, determination $G_{c.y}$ and G_T directly at the very beginning of design is impossible, since for this it is necessary to know value of weight G_0.

For determining weight of structure of aircraft G_F it is also necessary to know G_0, since sufficiently precision determination of weight of structure G_F possibly only then, when basic dimensions of aircraft are known; whereas sizes/dimensions can be determined only on the basis of weight G_0. Therefore in the first approximation, gross weight it is best to determine from the equation of the over-all payload ratios:

$$\overline{G}_{\kappa}+\overline{G}_{c.y}+\overline{G}_{T}+\frac{G_{c.з.г}}{G_0}=1,$$

where $G_{c.з.г}=G_{o6.п}+G_{r.r}+G_{c.н.ж}$ — sum, determined sufficiently accurately on the basis of the lists of equipment, catalogs and data of statistics, moreover $G_{п.н}$ and $G_{c.н.ж}$ — are assigned; $\overline{G}_{c.y}$ and $\overline{G}_T$ are determined from formulas. Determination $\overline{G}_{F}$ is possible only when in the first approximation, gross weight of aircraft is known. Calculation is conducted as follows.

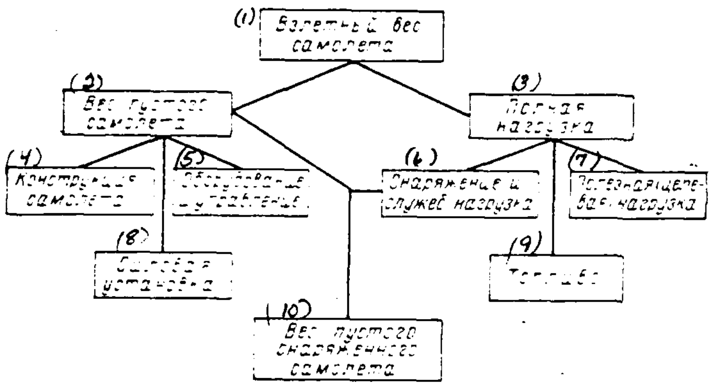

Fig. 8.1. Layout chart of gross weight of aircraft G_0 to components. Key: (1). The takeoff weight of aircraft. (2). Empty weight. (3). Full load. (4). Construction of aircraft. (5). Equipment and control. (6). Equipment and service load. (7) Useful (purposeful) load. (8). Power plant. (9). Fuel/propellant. (10). Weight of empty equipped aircraft.

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According to formulas (2.21) and (2.23) they determine $\bar{G}_{с.у}$ and $\bar{G}_{т}$, then according to formula (2.26); (2.26') or (2.26''), after assigning the probable value of gross weight of aircraft G_0' (using statistics), is found the value of the over-all payload ratio of construction/design in the first approximation, $\bar{G}_{к}'$. Since weight $G_{с.з.г}$ is known, they obtain $G_{с.з.г}/G_{с}'$ and find the sum

$$\bar{G}_{к}+\bar{G}_{с.у}+\bar{G}_{т}-G_{с.з.г}=\Sigma\bar{G}',$$

which will be more or less than one (or it is equal to one). If sum will be equal to one, then G_0' will be the first approximation of the unknown value of gross weight of aircraft. If sum is not equal to one, they are assigned by the second value of gross weight G_0'', they find $\overline{G}_{\cdot}$ and $G_{c.s.r}/G_0$ and they count sum $\overline{G}_{\cdot}-\overline{G}_{c.\cdot}-\overline{G}_{\tau}+G_{c.s.r}/G_0=\Sigma\overline{G}''$. Further is constructed dependence $G_0=f(\Sigma\overline{G})$ (Fig. 8.2), will be plotted points with coordinates G_0', $\Sigma\overline{G}'$, G_0'', $\Sigma\overline{G}''$ and is carried out through these points smooth curve. Intersection with curve with axis/axle G_0 gives the value of gross weight of aircraft in the first approximation, G_0^1.

For determination $\overline{G}_{c.y}$ it is necessary to know required thrust-weight ratio $\overline{P}_0$ and specific weight/gravity SU. The required thrust-weight ratio $\overline{P}_0$ is defined so, as noted above. Thew specific weight of the power plant r_0 can be determined for TRD according to the formula

$$r_0=\gamma_{\text{дв}}+\Delta\gamma_{\text{в}}, \quad (8.12)$$

where $\Delta\gamma_{\text{в}}=\varphi\overline{G}_{\tau}/\overline{P}_0$ ($\varphi=0.09$ for the small aircraft with the fuselage tanks; φ=0.13 for the large aircraft with the wing tanks);

$\overline{G}_{\tau}$ and $\overline{P}_0$ are taken according to the statistics or the calculation given above;

$\gamma_{\text{дв}}$ -- weight per horsepower can be accepted according to static data ($\gamma_{\text{дв}}=0.16-0.20$).

Over-all payload ratio of fuel/propellant $\overline{G}_{\tau}$ is determined in accordance with fact, is prescribed/assigned duration of flight t ' or distance L_{max} — according to formulas (2.22) or (2.23) and (2.23'). The entering the formulas coefficients ξ and ψ are determined according to the appropriate graphs. Coefficient c_e depending on velocity can be undertaken from the testing of model in wind tunnel of aircraft similar to that projected/designed or is calculated from the approximation formulas. Coefficient c_x is equal to $c_x = c_{x_0} + D_0 p_0^2 q^2$. For the aircraft with the piston and turboprop engines the over-all payload ratio of fuel/propellant is determined from formula (2.23'').

Frequently engine for projected/designed aircraft is prescribed/assigned. Then in the beginning of design are accurately known $\gamma_{дв}, c_{x_0}$, function $\xi = f_1(M)$ and $\psi = f_2(M)$ and task is reduced to check by calculation in the first approximation, will be requirements for the flight characteristics carried out. Is not prescribed/assigned, then it is necessary to make selection from several those being adequate/approaching. For the solution in the first approximation, of a question about the satisfaction of requirements on M_{max} are constructed the plotted functions $c_x(M)$ and $c_P(M)$. Thrust coefficient is determined from the formula

$$c_P = \frac{P_{Hi} n_{дв} p_0}{G_0^1 q},$$

where P_{Hi} — engine thrust at rated altitude H with given Mach number of flight;

$n_{дв}$ — quantity of engines;

G_0^1 — gross weight of aircraft in the first approximation;

q - velocity head.

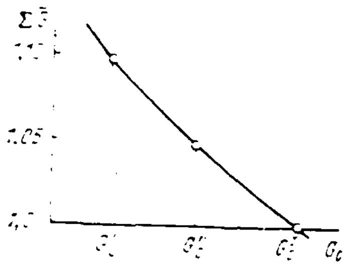

Fig. 8.2. Graphic method of determining gross weight of aircraft G_0 during sketch design.

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Through characteristics of selected engine are found weight of power plant $G_{cy}=P_{0i}n_{дв}\gamma_c$ (P_{0i} — boost for launching of engine) and fuel load

$$G_т \approx G_0^i \left(SR\beta c_{p0} \frac{L_{max} \sqrt{c_{x_0} D_0}}{M_{крейс}} + u \right),$$

where βc_{p0} — specific fuel consumption per $H_{крейс}$ and $M_{крейс}$, c_{x_0} and D_0 - for $H_{крейс}$ and $M_{крейс}$ (see Chapter II).

Value of number $M_{крейс}$ is determined graphically the structure in coordinates $M_{зад}$ and $M_{р}$ by curve $M_{крейс}$, counted according to formula $M_{крейс}=0.012\sqrt{\frac{P_0 n_{дв}\xi}{C c_x}}$, and ray/beam from origin of coordinates on an angle of 45° to axes/axles (see Fig. 2.17). By intersection with curve and ray/beam will be determined value $M_{крейс}$ at cruising altitude $H_{крейс}$, to which it will correspond

$$\Delta_{крейс} = \Delta_{вб} = \frac{1{,}760 G_0^i \sqrt{c_x D_0}}{P_{0i} n_{дв}\xi},$$

where $P_{\text{сгь}}\xi$ — thrust of selected engine on $M_{\text{крейс}}$ according to engine characteristic for H=0 (coefficient ξ calculates change in thrust on the basis of velocity, and also thrust losses in input devices SU with M>1).

Having defined value $G_{c.y}$, G_{T} and $\overline{G}_{K}$, gross weight in second approximation/approach

$$G_0^{II}=\frac{G_{c.э.r}-G_{c.y}-G_{T}}{1-\overline{G}_{K}}. \tag{8.13}$$

is determined then value $S=G_0^{II}\,p_0$ is found.

Knowing S, they begin layout of aircraft, they choose and make more precise all sizes/dimensions and parameters of aircraft components, they develop/process and make more precise its general view and calculate weight of structure of aircraft components, using for this weight formulas (for wing, fuselage, tail assembly, chassis/landing gear, etc.). After this, gross weight in the third approximation/approach is determined:

$$G_0^{III}=G_{c.э.r}+G_{c.y}+G_{T}+G_{K}$$

they further compose the combined weight (see Appendix I).

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Chapter IX.

SPECIAL FEATURES OF THE DESIGN OF PASSENGER AIRCRAFT.

§ 1. Layout of passenger compartment.

With layout of passenger aircraft it is important to ensure arrangement/position on aircraft of given number of passengers (payload) with smallest empty weight. First of all the solution of this problem is connected with the selection of the optimum sizes/dimensions of fuselage, passenger compartment and auxiliary locations.

At present in practice of air passenger transportation are used three different classes of cabins/compartments, which differ from each other in terms of comfort of arrangement/position of passengers and in terms of conditions of care of passengers in flight.

In highest, I class, is increased width of seats and distance between rows (space of seats), passengers more frequently obtain feed/supply (is increased space of kitchens), etc.

In II, either tourist, and III, or economic, classes respectively narrower seats and smaller space between rows of seats.

Sizes/dimensions of passenger compartment are determined by number of passengers, by construction/design and sizes/dimensions of passenger seats.

Basic dimensions of passenger seats are given in following table (Fig. 9.1).

Классы пассажирских сидений (1)	Расстояние между подлокотниками B (2)	Ширина подлокотника b (3)	Длина подушки сидения до спинки L (4)	Ширина подушки сидения B_1 (5)	Высота сидения над полом h (6)	Ширина спинки B_2 (7)	Длина спинки от подушки сидения l_1 (8)	Угол отклонения спинки от вертикали β (9)	Высота сидения H (10)	(11) Ширина блока сидений B_3	B_4
I класс	470	70—80	470	—	300	—	720	55—65	1100	1200	1420
Туристский	440	50—60	450	430	320	430	700	35	1100	1030	1520
Экономический	420	40—50	430	—	320	—	700	25	1100	970	1430

All sizes/dimensions are given in mm.

KEY: (1). Classes of passenger seats. (2). Distance between elbow-rests. (3). Width of elbow-rest. (4). Length of pad of seat to back. (5). Width of pad of seat. (6). Height/altitude of seat above floor. (7). Width of back. (8). Length of back from pad of seat. (9). Angle of deflection of back from the vertical line. (10). Height/altitude of seat. (11). Width of the block of seats. (12). class. (13). Tourist. (14). Economic.

It is necessary to keep in mind that:

a) passenger seats usually are done in the form of blocks of two or three seats. For the I class only the blocks with two seats are used; for the II and III classes are possible the blocks both with two and with three seats;

b) seats of the I class must have a back, which is removed back/ago to the angle to 55°-65° from the vertical line, and removable area/site - pad in order to ensure reclining the position of passenger for rest;

c) seats of the II and III classes must have a back, which is freely removed forward so that with the emergency the passenger of rear row could not be hammered by head against the back of front seat;

d) seats of the III class due to the low pitch between the seats must be hinged/reversible;

e) each seat of the I class compulsorily must have two elbow-rests with the width not of less than 70 mm. In the blocks of seats of the II and III classes average/mean elbow-rests are done on one among the seats; their width comprises respectively not less than 60 mm and not less than 40 mm;

f) during the installation in one row of several blocks of seats the width of the passage between these blocks must be:

for the I class - not less than 500 mm;

for the II class - 450-500 mm;

for the III class - 390-400 mm.

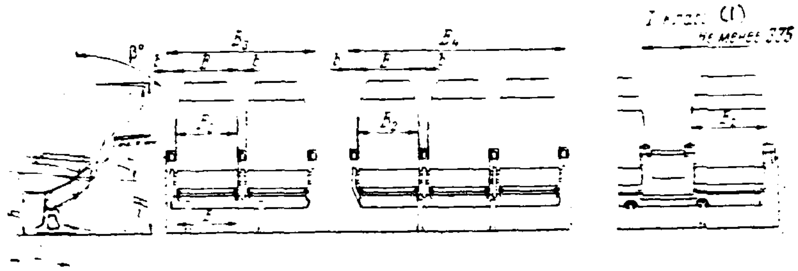

Fig. 9.1. Basic dimensions of passenger seats.

Key: (1). Class is not less.

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Weight characteristics of seats are given in Table 9.1.

If we consider that distance from elbow-rest of seat to internal surface of fuselage must be not less than 50-60 mm and wall thickness of fuselage is usually 120 mm (construction/design, heat and sound insulation, decorative coating), then it is easy to calculate required width of fuselage in zone of arrangement/position of seats.

For example, for positioning/arranging six seats of II class in row (two blocks on three seats) required width of fuselage (or diameter of fuselage with round cross section)

$$B_d = 440 \cdot 6 + 50 \cdot 8 + 400 + 60 \cdot 2 + 120 \cdot 2 = 3800 \text{ мм}^{(1)},$$

Key: (1). mm.

where 440 - width of seats;

50 - width of elbow-rests;

400 - width of passage;

60 - clearance to the walls;

120 - thickness of walls.

The calculations, made similarly for different number of seats in the row, they are given in Table 9.2.

Sizes/dimensions of passenger compartment are determined in essence by number of passengers. On a number of passengers in one row and, consequently, the specific width of passenger compartment and the width of fuselage is determined a required number of rows of seats.

Table 9.1.

Класс сидений (1)	Вес блока из двух сидений в кгс (2)	Вес блока из трех сидений (3) в кгс
I	22—30	—
II	18—25	27—35
III	15—20	22—30

Key: (1). Class of seats. (2). Weight of double in kg. (3). Weight of block of three seats in kg.

Table 9.2. Minimally necessary width of fuselage.

Число сидений в ряду (1)	Количество и тип блоков (2)	Ширина прохода в мм (3)	Ширина и количество подлокотников в ряду (4)	Зазор между внешним подлокотником и стенкой (5)	Ширина фюзеляжа в зоне пассажирских сидений или диаметр фюзеляжа в мм (6)
3	1+2	390	50 (2+3)	50×2	2300
4	2+2	400	50 (3+3)	50×2	2800
5	2+3	400	50 (3+4)	50×2	3300
6	3+3	400	50 (4+4)	60×2	3800
7	2+3+2	480+480	50 (3+4+3)	60×2	4900
8	2+3+3	485—485	50 (3+4+4)	60×2	5400
9	3+3+3	490+490	50 (4+4+4)	60×2	5900
10	3+2+2+3	500+500	50 (4+3+3+4)	60×2+140	6600

Notes. 1. Width of the seat between the elbow-rests 440 mm; thickness of one wall of fuselage 120 mm. 2. with number of seats in row of 9 and layout on scheme 3+2+2+2 due to supplementary elbow-rest and clearance between average/mean blocks of seats width of fuselage increases to 6000 mm.

Key: (1). number of seats in the row. (2). Quantity and the type of blocks. (3). Width of passage in mm. (4). Width and a quantity of elbow-rests in the row. (5). Clearance between the external elbow-rest and the wall. (6). Width of fuselage in zone of passenger seats or diameter of fuselage in mm.

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Length of passenger compartment is located taking into account following conditions:

a) distance or space between rows of seats;

in I class - 960-1080 mm;

in II class - 840-870 mm;

in III class - 750-810 mm;

b) first row of seats must be placed not nearer than on 1200-1300 mm from front partition of cabin/compartment, if we count this size/dimension from plane of partition to top of back of seat in its normal position (angle of slope of 15- 18°) as this shown in Fig. 9.2.

If first row of seats is placed by backs on flight (passengers they sit facing tail of aircraft), then second row of seats must be found not nearer than on 2000-2200 mm from front partition as this shown in Fig. 9.3. This layout is advisable in all cases, if in the cabin/compartment there is no such means of entertainment as cinema; the passengers of the first row in this case do not tire in flight the type of the closely spaced partition;

c) for the possibility of deviating the back of seat of the maximum permissible angle between the top of the back of the seat of rear row and the plane of rear partition the distance must be not less than 235-250 mm in layouts I and the II classes and not less than 35-50 mm in layout of the III class (see Fig. 9.2).

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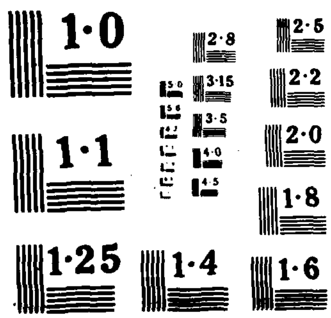
1·0
2·8
2·5
3·15
2·2
3·5
1·1
2·0
4·0
4·5
1·8
1·25
1·4
1·6

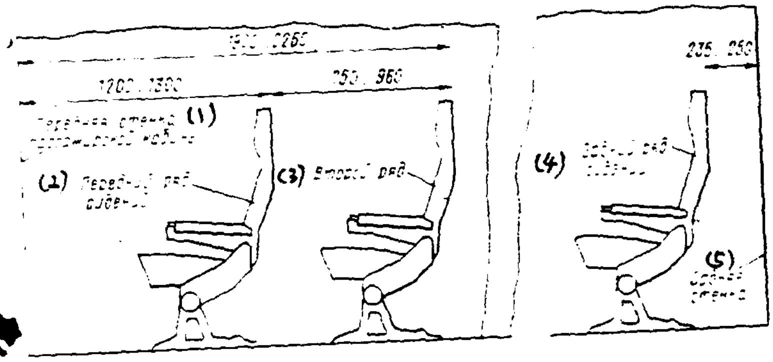

Fig. 9.2. Diagram of installation of passenger seats relative to front and rear partitions of cabin/compartment.

Key: (1). Front wall of passenger compartment. (2). Front row of seats. (3). Second row. (4). Rear row of seats. (5). Rear wall.

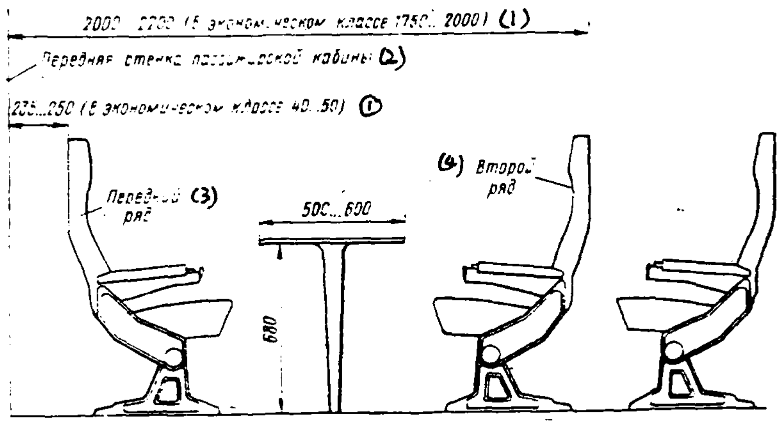

Fig. 9.3. Version of arrangement of front rows of passenger seats against each other.

Key: (1). In the economic class. (2). Front wall of passenger compartment. (3). Front row. (4). Second row.

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Let us examine two examples.

1. It is necessary to find length of passenger compartment, if number of passengers 150, they are placed on 6 people in row and layout must correspond to III class. The length of cabin/compartment will be, obviously, is equal to

$$l_{каб} = 1200 + \left(\frac{150}{6} - 1\right)780 + 50 = 19\,970 \text{ мм}^{(1)} \text{ или } \sim 20\,000 \text{ мм},$$

Key: (1). mm or ~20000 mm.

where 1200 - distance from the partition to the first row in mm;

780 - space between the rows of seats in mm;

50 - distance from the partition to the latter/last row in mm;

$\left(\frac{150}{6}-1\right)$ - number of gaps/intervals between the rows.

2. How many passengers it is possible to place in the same cabin/compartment of II class, by accepting that they also are placed on 6 people in row?

$$n_{\text{пас}}=6\left(\frac{20\,000-1260-260}{840}+1\right)=6\cdot 23=138 \text{ чел.,} (1)$$

Key: (1). man.

where 20000 - length of cabin/compartment in mm;

1260 - distance from the front partition to the top of the backs of the first row of seats in mm;

260 - distance between the top of the backs of the last row of seats and the rear partition in mm;

840 - space between the rows of seats in mm.

Selection of sizes/dimensions of passenger compartment is checked by obtained space of this cabin/compartment or, it is more precise, with specific volume of this cabin/compartment, which falls to one passenger. This specific volume:

for layout of the I class ≥1.5-1.8 m^3/man;

for layout of the II class ≥1.2-1.3 m^3/man;

for layout of the III class ≥0.9- 1.0 m^3/man.

In this case it is necessary to keep in mind that the greater flying range of aircraft, the more must be specific volume of passenger compartment and that into space of passenger compartment space of vestibules, cloakrooms, dress/lavatories and kitchens does not enter.

Height/altitude of passenger compartment in zone of passage between seats must be not less than 1900-2000 mm.

§ 2. Layout of official compartment and operating locations.

Official cabin/compartment or flight deck is placed in forward fuselage section. Its sizes/dimensions and layout depend on the composition of crew. Usually crew consists of four people: the first pilot - craft commander, co-pilot, flight engineer and navigator. The possible versions of the arrangement/position of this crew in the cabin/compartment are shown in Fig. 9.4.

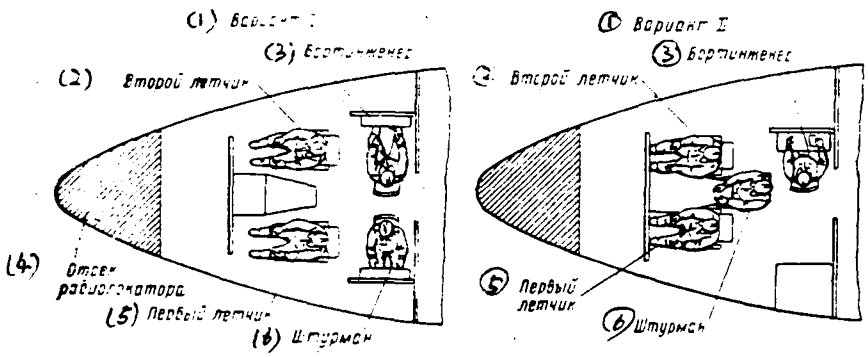

Fig. 9.4. Possible diagrams of arrangement/position of crew in official cabin/compartment.

Key: (1). Version. (2). Co-pilot. (3). Flight engineer. (4). Section of radar. (5). First pilot. (6). Navigator.

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In passenger aircraft, intended for operation on average/mean and short routes, equipped with ground- based radio beacons, composition of crew usually decreases to three people due to removal/taking of navigator. In the long- range main-line passenger aircraft, vice versa, the composition of crew can increase due to the introduction to it of special radio operator.

In passenger aircraft for feeder lines crew usually consists only of two pilots.

In official cabin/compartment, as a rule, one additional or two spare places with seat belts for positioning/arranging on them of instructors, inspector or navigator-pilot are provided for.

By most important element of layout of official cabin/compartment is arrangement/position of pilots and guarantee it of necessary survey/coverage.

Standard layout of place of pilot is shown in Fig. 9.5 and 9.6. Requirements for the survey/coverage of pilot are given below.

1. Zone of unimpeded azimuth coverage:

to the left - 20°;

to the right - from 20° to 30°;
downward - not less than 16°;
upward - 10°-20°.

2. In the range between 20°-45° of left side installation of one power strut is allowed/assumed.

3. Coverage back must provide visibility of half of end fairing about outer plane of wing.

4. Width of shading/blanketing fundamental power struts - is not more than 70 mm.

5. Size/dimension $L_c = 500$ mm and more with guarantee of viewing angle is not less than 16° down and not less than 20° upward.

6. During removal/taking of diagram of survey/coverage from point C_1 displacement of head (in limits of drawing of seat belts) over R=100-120 mm is allowed/assumed.

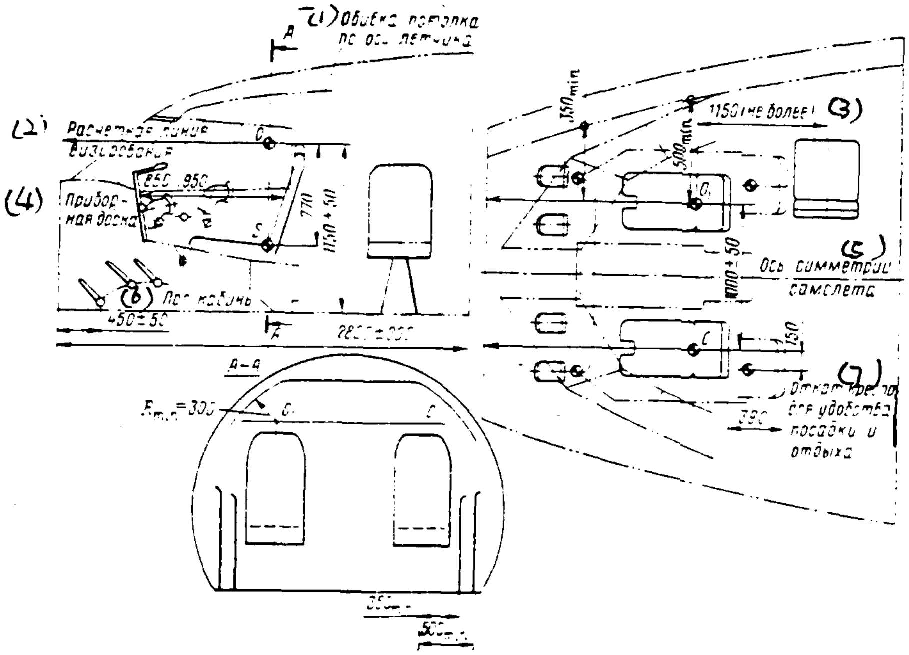

Fig. 9.5. Diagram of arrangement/position of pilots in passenger aircraft with control panel between pilots.

Key: (1). Casing ceiling along the axis/axle of pilot. (2). Reference line of sighting. (3). Not more. (4). instrument panel. (5). Axis of symmetry of aircraft. (6). Floor of cabin/compartment. (7). Recoil/backsweep of seat for convenience in landing and rest.

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In composition of crew for maintaining order in passenger location, preparation and distribution of feed/supply to passengers in flight, instruction on use of one or the other means of passenger equipment and by emergency systems and rendering to first aid enter stewards.

Their number in aircraft is determined from following conditions:

a) two stewards to department/separation of I class with number of passengers to 30;

b) one steward to 50 passengers in cabins/compartments of II and III class.

Each steward must compulsorily occur with seat belts; it can be hinged/reversible.

In aircraft must be cloakroom for outer clothing of stewards and air crew.

For positioning/arranging luggage of passengers in passenger aircraft baggage holds are provided for. They, as a rule, must be located under the floor of passenger compartment or in the ground floor and have the height not less than 1100 mm. The space of baggage holds is selected so that in them it would be possible to transport, besides the standardized luggage of passengers (20 kg to each passenger with the specific weight/gravity of luggage 120 kG/m³), mail and urgent load. It is usually accepted that the necessary space of the baggage holds

$$v_{баг} \approx (0.23-0.25)\, n_{пас}\ м^3, \qquad (9.1)$$

where $n_{пас}$ - number of passengers.

With diameter of fuselage less than 2800 mm, baggage holds under floor be located cannot and then they place from behind passenger location from the front and so that, regulating their load, it would be possible to retain center-of-gravity location of aircraft (centering) within prescribed/assigned limits.

For guaranteeing loading luggage and mail in standard containers luggage hatches must have sizes/dimensions, indicated in Fig. 9.7.

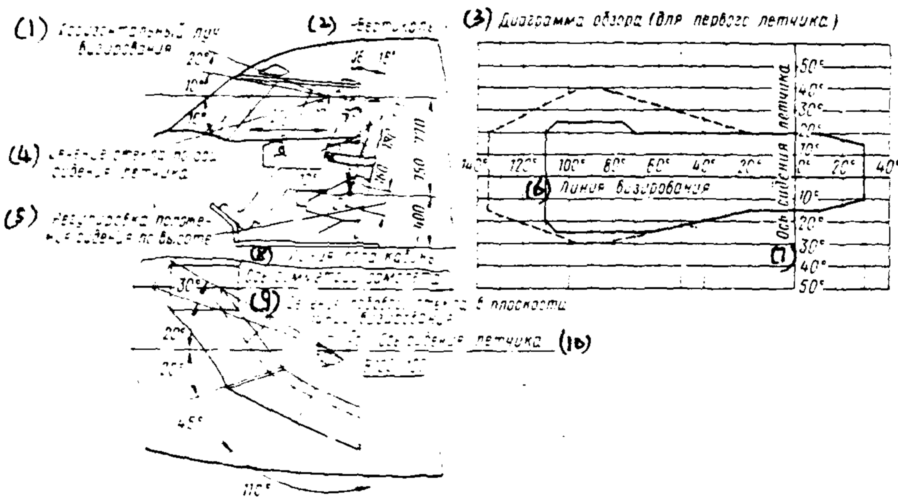

Fig. 9.6. Required by the norms of the airworthiness of passenger aircraft view from the cabin of the pilot: C_1 - position of the eye of pilot during takeoff and landing; C_2 - position of the eye of pilot in flight of aircraft; ——— the zone of necessary survey/coverage; - - - the zone, desirable for the survey/coverage.

Key: (1). Horizontal ray/beam of sighting. (2). Vertical line. (3). Diagram of survey/coverage (for first pilot). (4). Cross section of glass along the axis/axle of the seat of pilot. (5). Regulation of position of seat on height/altitude. (6). Line of sighting. (7). Axis/axle of the seat of pilot. (8). Line of floor of cabin/compartment. Axis of the symmetry of aircraft. (9). Cross section of frontal glass in the plane of the line of sighting. (10). axis/axle of the seat of pilot.

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Baggage holds compulsorily must be located in the zone of pressurized cabin.

For guarantee with nourishment of passengers in flight in passenger location are provided for one or two kitchens, into which on airfield are charged containers with packed food and beverages. In flight the stewards only heat the necessary quantity of food and resound or struggle on the special trolleys the served trays to the passengers, in whom on the seats, which stand in front, are hinged/reversible stands.

One kitchen in aircraft is done in such a case, when there is same-type layout of passenger compartment and number of passengers does not exceed 100 people. If in passenger compartment are passenger cabins, then independent of a number of passengers in the aircraft must be two kitchens, whose total volume comprises

$$v_{\text{кух}} \approx 0.12-0.14\ n_{\text{ас}}\ \text{м}^3. \qquad (9.2)$$

Quantity of toilet locations in passenger aircraft depends on number of passengers and on duration of flight. With the duration of flight of less than two hours should be made one dress/lavatory to 50 passengers; with the duration of flight from two to four hours - one dress/lavatory to 40 passengers with a number of passengers 120 and less, one dress/lavatory to 45-50 passengers with a number of passengers is more than 120; with the duration of flight it is more than four hours - one dress/lavatory to 30 passengers with their number less than 120, to 40 passengers - with their number from 120 to 200 and to 45-50 passengers - with a number of passengers it is more than 200.

Floor space of toilet location must be 1.5-1.6 m^2, width - is not less than 1 m.

Problem of cloakrooms is most indefinite in layout of passenger aircraft. It is urgent/actual only for the aircraft, which are operated on the air lines of the USSR, Canada, Alaska and the Scandinavian countries in the winter time; in the summer time in these countries and in all seasons in the remaining countries of coat and the head-gear can be placed on the luggage shelves.

It is required so that for cloakrooms location with floor space of

$$S_{гард}=(0{,}035-0{,}050)\,n_{пас}\ м^2. \tag{9.3}$$

would be abstracted/removed and space of this location must comprise

$$v_{гард}\leqslant(0{,}05-0{,}08)\,n_{пас}\ м^3. \tag{9.4}$$

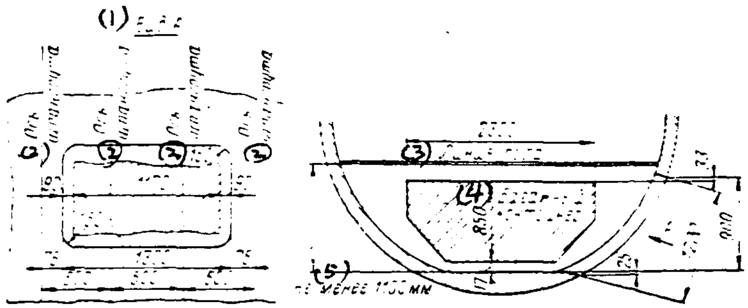

Fig. 9.7. Basic recommended dimensions of luggage hatches.

Key: (1). form. (2). Axis of frame/former. (3). Line of floor. (4). Luggage container. (5). Not less than 1100 mm.

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Organization of cloakrooms with removable partitions is most rational solution of this problem so that in summer time instead of cloakrooms in aircraft it would be possible to establish/install supplementary passenger places. Therefore cloakrooms should be placed either in the vestibules at the entrance doors or in the zones, where the noise within passenger compartment is greatest.

§ 3. Provision of emergency evacuation of passengers. Doors and windows in the passenger location.

During design of passenger aircraft must be provided possibility:

- emergency evacuation of all passengers and crew on earth/ground during 90-120 seconds in released and gear up position;

- emergency debarkation of passengers into onboard emergency floating means (inflatable rafts) during forced landing on water surface.

For exit of passengers under conditions of urgent evacuation on earth/ground from each side of fuselage emergency exits (Table 9.3) must be created.

I type hatch - sizes/dimensions in light/world 610×1220 mm; hatch is arranged/located at the level of floor of cabin/compartment out of zone of wing.

II type hatch - sizes/dimensions in the clear 510×1120 mm; hatch is arranged/located either at the level of floor out of zone of wing, or in zone of wing, having the height of lower edge from level of floor of cabin/compartment not more than 250 mm and from wing surface not more than 430 mm.

III type hatch - sizes/dimensions in the clear 510×915 mm; hatch is arranged/located in zone of wing, having the height of lower edge from floor level not more than 510 mm and from wing surface not more than 685 mm.

IV type hatch - sizes/dimensions in the clear 480×700 mm; hatch is placed in zone of wing, having the height of lower edge from floor level not more than 735 mm and from wing surface not more than 915 mm.

By dimensions in the clear are understood sizes/dimensions of door opening in plane of its chord.

Note. 1. Corners of hatches must be rounded off with radius not of more than 1/3 widths.

2. Main entrance doors are considered in number of I type emergency exits.

3. With increase two times of width of I type emergency exits for provision of simultaneous passage through it of two people total number of emergency exits can be reduced to 1/3.

4. On aircraft with upper wing arrangement, besides indicated in Table 9.3 number of ejection hatches from each side of fuselage, must be emergency exits of III type above of fuselage from calculation 1 hatch to 35 passengers.

In immediate proximity of emergency exits I and II rubber ladders, which automatically inflate with ejection (Fig. 9.8), or tightening linen (aircraft tarp) grooves.

Table 9.3.

Количество пассажиров (1)	Люк I типа (2)	Люк II типа (2)	Люк III типа (2)	Люк IV типа (2)	Количество пассажиров (3)	Люк I типа (2)	Люк II типа (2)	Люк III типа (2)	Люк IV типа (2)
7—19	—	—	1	—	140—179	2	—	2	—
20—39	—	1	—	1	180—219	2	2	—	—
40-59	1	—	—	1	220—249	3	—	2	—
60—79	1	—	1	—	250—279	3	2	—	—
80—109	1	—	1	1	280—309	4	—	2	—
110—139	2	—	1	1					

Key: (1). quantity of passengers. (2). Hatch ... of the type. (3). Quantity of passengers.

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For providing of safety of passengers and crew in flight above sea (with flight time from coast seaward of more than 90 min) must be provided for rescue inflatable rafts (boats) in quantity and total load capacity, which correspond to number of passengers and crew.

Normal entry of passengers into cabin/compartment is accomplished/realized through doors, arranged/located, as a rule, on left side (on aircraft with number of passengers of more than 250 doors they can be placed on both boards). Door opening is from below limited to the plane of floor. Threshold is not allowed/assumed. The sizes/dimensions of door openings are indicated in Fig. 9.9.

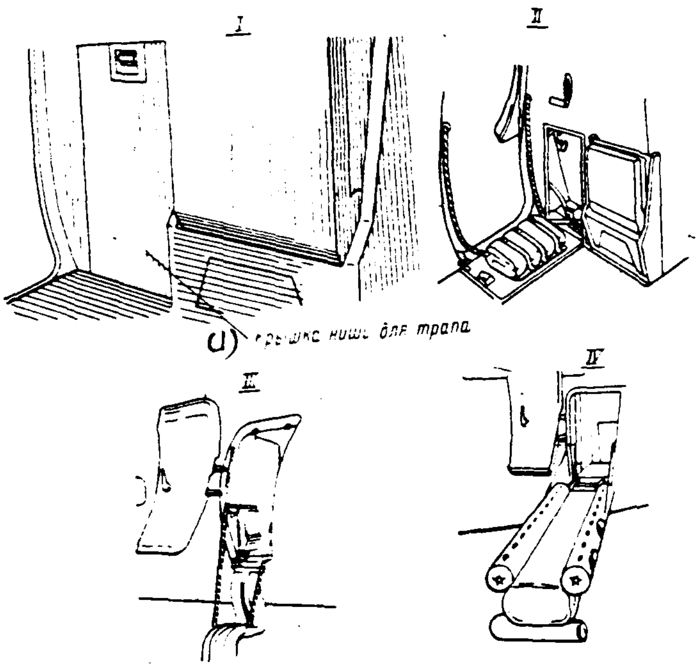


Fig. 9.8. Arrangement/position of emergency inflatable chute: I - flight position (cover/cap of bay for ladder it is closed); II - ladder in position "is ready for jettisoning"; III -door is opened, ladder is rejected/thrown out; IV - ladder is automatically inflated and ready for descent of passengers.

Key: (1). cover/cap of bay for the ladder.

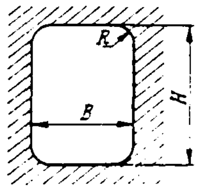
Диаметр (высота) фюзеляжа (1)	H	B	R	Форма двери (4)
	(2) не менее		не более (3)	
	мм			
(5) от 2000 до 2500 (6)	1400	850	1/3B	
(5) от 2500 до 3500 (6)	1600			
свыше 3500 (7)	1800			

Fig. 9.9. Recommended sizes/dimensions of openings of entrance doors. Key: (1). diameter (height/altitude) of fuselage. (2). Not less. (3). Not more. (4). Form of door. (5). from. (6). to. (7). It is more than.

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Doors must be opened/disclosed outside so that in emergency situation passengers would not prevent their opening. It is at the same time desirable so that the closed doors would be forced by internal pressure in the cabin/compartment against edging door opening, which prevents/warns the spontaneous opening of doors in flight during the disturbance/breakdown of efficiency of locks.

Windows of passenger compartment must be continuous light line. Structurally this is fulfilled in the form of the separate small windows (width of 200-230 mm, the height/altitude of 320-350 mm), situated between the frames/formers (space of windows 500-510 mm). With this construction/design the possibility of the survey/coverage through the windows is provided with any space of the seats of passengers all the more that in the practice of operation changes of the layouts of passenger compartment with the appropriate change in the space of seats are possible.

§ 4. Selection of the basic dimensions of the wing of subsonic passenger aircraft.

Wing area of subsonic passenger aircraft is selected, on the basis of guarantee of flight conditions at base altitude, that corresponds to maximum lift-drag ratio, and also from guarantee of prescribed/assigned approach speed.

1. From first condition

$$S = \frac{G_{\text{cr}}}{q c_{y\,\max\,\text{крейс}}}\ \text{м}^2, \qquad (9.5$$

where G_{cr} - average weight of aircraft on route in kg;

q - velocity head at base altitude and at prescribed/assigned flight speed in kgf/m²;

$c_{y\,\max\,\text{крейс}}$ - value c_y, which corresponds to maximum value $K_{\text{крейс}}$,

$$c_{y\,\max\,\text{крейс}} = \sqrt{\frac{c_{x_0} - \frac{c'_x F}{S}}{D_0}}, \qquad (9.6)$$

where F - areas of midsection of noncarrying elements of aircraft (fuselage, nacelles, engines, etc.);

c'_x - coefficient of aerodynamic drag of any aircraft component, pertaining to appropriate area of midsection;

$D_0 = \frac{dc_x}{dc_y^2}$ - coefficient of refuse/bank of polar in formula

2. From second condition

$$S = \frac{27 G_{\text{пос}}}{V^2_{\text{з.пос}} c_{y\,\max\,\text{пос}}}\ \text{м}^2, \tag{9.7}$$

where $G_{\text{пос}}$ - landing weight of aircraft in kg;

$V_{\text{з.пос}}$ - prescribed/assigned approach speed in m/s;

$c_{y\,\max\,\text{пос}}$ - maximum value c_y in landing position of high-lift device of wing.

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During design of aircraft for design wing area the larger of the values, determined of two conditions, indicated above is accepted. In order to obtain aircraft with the minimal sizes (and cc ently, and most economical) according to formulas (9.5) and (9.7), it is necessary to have:

- the largest possible value c_y with the maximum value of K, i.e., the coefficient of refuse/bank of polar D_0 must be minimum. Coefficient D_0 is directly proportional to losses to balancing/trimming of aircraft and it is inversely proportional to wing aspect ratio;

- the largest possible value $c_{y\max}$ in the landing position of the high-lift device of wing. An indispensable condition of this is the mechanization/lift-off device not only of trailing edge (flaps), but also leading edge (slot slats).

For passenger aircraft of large and medium range with three engines and more selection of sweepback of wing is uniquely determined by assigned value of cruising flight speed. Since on the condition of obtaining the greatest efficiency/cost- effectiveness the value of cruising speed must be as large as possible, also, at the same time aircraft simultaneously it must have maximally possible lift-drag ratio, then according to the data of Fig. 9.10 and 9.11 it follows that the subsonic aircraft can have the cruising flight speeds, which correspond to number M=0.85-0.90, and the sweepback of wing in this case along the line of 1/4 chords must be

$$\chi = 35^\circ - 40^\circ.$$

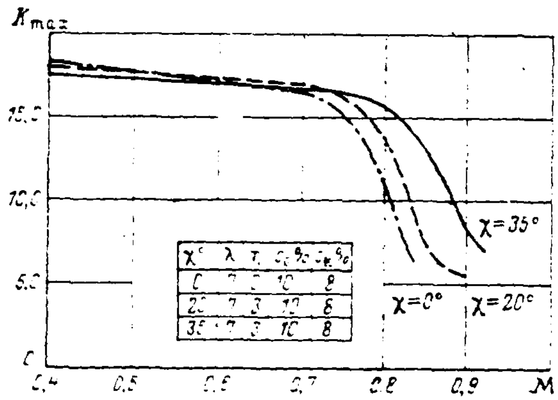

Fig. 9.10. Change of lift-drag ratio of wing K_{max} in dependence on sweep angle of wing χ and on Mach number of flight.

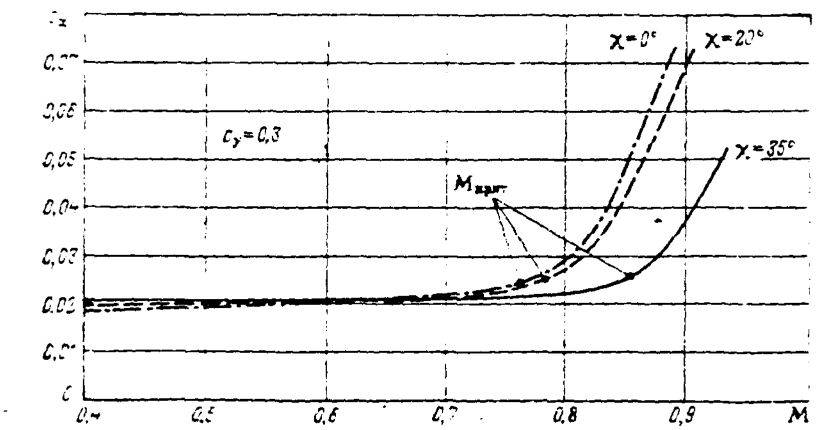

Fig. 9.11. Effect of sweep angle of wing χ on value $M_{крит}$.

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For passenger aircraft with low flying range, when high values of cruising speed cannot be completely realized and especially for passenger twin-engine aircraft, on selection of optimum sweepback is placed supplementary condition: guarantee of continuation of takeoff of aircraft in the case of failure of one engine with computed values of temperature and pressure of surrounding air (+30°C; 730 mm Hg).

According to norms of airworthiness of commercial airplanes of USSR [1] angle of slope of trajectory of climb of aircraft in the case of failure of one (most critical) engine must be not less:

Количество двигателей на самолете (1)	2-й этап (2)	3-й этап (3)	4-й этап (4)
2	0,5%	2,5%	1,1%
3	1,1%	2,7%	1,4%
4	1,3%	3,0%	1,5%

Key: (1). quantity of engines on aircraft. (2). 2nd stage. (3). 3rd stage. (4). 4th stage.

FOOTNOTE [1]. Norms of the airworthiness of civil/civilian aircraft of the USSR. MAP USSR, MGA, 1967. ENDFOOTNOTES.

Note. The 2nd stage of takeoff - from the height/altitude of 10 m above the end/lead of runway (above the end/lead of KPB - clear zone) to the end/lead of the landing gear retracting; the high-lift device of wing in the takeoff position; the length of the section of 2 stages - is 300 m.

3rd stage of takeoff - from end/lead of 2 stages to height/altitude of 100 m above level of flight field; landing gear is retracted, high-lift device of wing in takeoff position.

4th stage of takeoff - from height/altitude of 100 m to height/altitude of 400 m; high-lift device of wing is removed since beginning of 4th stage of takeoff.

Flight path angle of climb is determined by expression

$$\operatorname{tg}\theta = \frac{P_{\text{вз}}}{G_{\text{вз.л}}} - \frac{1}{K_{\text{вз.л}}}, \qquad (9.8)$$

where $P_{\text{вз.л}}$ - gross thrust of engines on aircraft with one failed engine at rate of climb and in those atmospheric conditions, which are accepted calculated for this airfield in kg;

$G_{\text{вз.л}}$ - takeoff weight of aircraft in kg;

$K_{\text{вз.л}}$ - aerodynamic aircraft quality/fineness ratio under the conditions, which corresponds to rate of climb and to given atmospheric conditions.

In 2nd stage $K_{\text{вз.л}}$ it is accepted taking into account of high-lift device of wing in takeoff position and released, but gradually retractable landing gear; in 3rd stage $K_{\text{вз.л}}$ it is accepted taking into account only takeoff position of high-lift device of wing and in 4th stage - with retracted the landing gear and high-lift device of wing.

From formula (9.8) it is easy to see that nominal required thrust with all operating engines:

- for twin-engine aircraft

$$P_{\text{потр}} = 2G_{\text{вз.л}}\left(\frac{1}{K_{\text{вз.л}}} + \operatorname{tg}\theta\right); \qquad (9.9)$$

- for aircraft with three engines

$$P_{\text{потр}} = \frac{3}{2}G_{\text{вз.л}}\left(\frac{1}{K_{\text{вз}}} + \operatorname{tg}\theta\right); \qquad (9.10)$$

- for aircraft with four engines

$$P_{\text{потр}} = \frac{4}{3}G_{\text{вз.л}}\left(\frac{1}{K_{\text{вз.л}}} + \operatorname{tg}\theta\right). \qquad (9.11)$$

Since tgθ is prescribed/assigned to standard/normal of airworthiness, and $G_{вз.л}$ is initial parameter at investigation, then value of required thrust depends on aerodynamic aircraft quality/fineness ratio in takeoff position of high-lift device of wing.

Value $K_{вз.л}$ depends on hum of sweepback of wing, increasing with decrease of sweep angle as this shown in Fig. 9.12.

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Second factor, which affects value $K_{вз.л.}$ are structural-aerodynamic characteristics of high-lift device of wing, represented in Fig. 9.13.

Solving variational problem of selecting corresponding mechanization/lift-off device of wing and sweep angle for obtaining predetermined trajectory of climb on 3rd stage of takeoff and optimum climb on the 4th (climb it is steeper, the less effect of noise of aircraft on populated territories adjacent to airfield), is selected optimum sweep angle of wing. In this case it is necessary to keep in mind the first condition for the selecting of the sweepback of the wing: the guarantee as of large as possible cruising flight speed.

Sweepback of wing in row of twin- engine aircraft (BAC-111, Douglas DC-9, Sud Aviation "Caravelle", Boeing 737, etc.) composes

20°-25°.

Wing aspect ratio λ for subsonic passenger aircraft is designed taking into account fact that increase in elongation/aspect ratio favorably affects value of lift-drag ratio, but simultaneously it leads to gain in weight of wing, and consequently, and entire aircraft.

For wing with sweepback of χ=35° optimum is elongation/aspect ratio λ=6.0-7.5; for wing with sweepback of χ=20-25° optimum wing aspect ratio λ=7.0-8.0.

Wing taper in plan/layout (ratio of root wing chord to end) to a considerable extent determines stalling characteristics of wing high angles of attack and it affects characteristics of longitudinal and lateral stability of aircraft, impairing these characteristics with increase in contraction; on the other hand, increase in wing taper leads to reduction of its weight.

Optimum values of wing taper η lie/rest within limits

$$\eta=3,5-4,5.$$

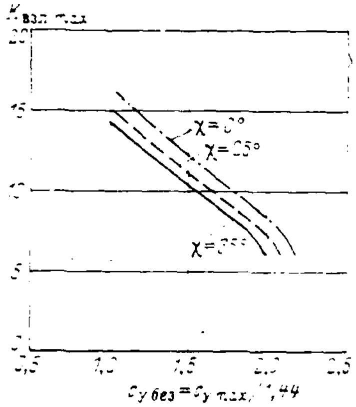
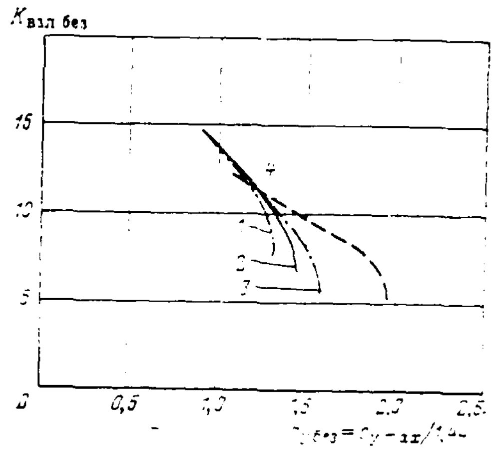

Fig. 9.12 Fig. 9.13.

Fig. 9.12. Change of value $K_{взл\ max}$ in dependence on $c_{y\ без}$ at different constant sweep angles (λ=9).

Fig. 9.13. Effect of type of high-lift device of wing on value K. 1 - single- slotted flap; 2 - double-slotted flap; 3 - three-slot flap; 4 -three-slot flap and slat.

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Selection of relative thickness of wing sections is closely related to sweep angle of wing, also, with airfoil characteristics of wing. For guaranteeing the smallest weight of the wing construction and obtaining of the greatest tank volume for the fuel/propellant are selected such maximum permissible values of thickness ratio of wing, with which the formation/education of supersonic zones the flows shifted large Mach numbers of flight or these zones are arranged/located so, that is formed supplementary suction force, which decreases c_x of wing.

Optimum average/mean relative thickness of cross sections (profiles/airfoils) of wing with sweepback of wing of χ=35° lies/rests within limits

$$\bar{c}=10-12\%,$$

providing in this case reaching/achievement of cruising flight speeds, which correspond to Mach number of flight, to equal to 0.85-0.90.

In wings with χ=20-25° average/mean relative thickness of wing sections is usually the same, taking into account that cruising speeds in aircraft with such wings less lie/rest within limits, which correspond to Mach numbers of flight, equal to 0.7-0.8.

§ 5. Selection of thrust-to-weight ratio and numbers of engines for the subsonic passenger aircraft.

Required thrust of all engines of passenger subsonic aircraft in level flight on to base altitude and flight speed is determined by formula

$$P_{\text{потр. крейс}}=\frac{G}{K_{\text{крейс}}}+c_x'Fq \text{ кгс}, \qquad (9.12)$$

where G - weight of aircraft in the beginning of cruising section of path in kg;

$K_{\text{крейс}}$ - lift-drag ratio of wing, which corresponds to flight conditions (in required value c_y).

F and c_x' - area of midsection of all noncarrying elements of aircraft and coefficient of drag, in reference to this area.

Obtained value $P_{\text{потр}}$, divide by number of engines, must correspond to cruise setting of work of engine, not limited on time of continuous operation.

On value of this thrust on to base altitude and flight speed according to altitude-speed engine characteristics required total static thrust of engines and thrust-weight ratio of aircraft in the form

$$\overline{P}_0=\frac{P_{0i}n_{\text{дв}}}{G_0}, \quad (9.13)$$

where $\overline{P}_0$ - thrust-to-weight ratio of aircraft, is determined;

P_{0i} - static engine thrust on takeoff mode in kg;

$n_{\text{дв}}$ - number of engines;

G_0 - takeoff weight of aircraft in kg.

In number of cases to passenger aircraft are presented supplementary requirements on completion of flight in the case of failure of one or two engines in any section of route, namely: aircraft with engines, which preserved efficiency and which work in nominal rating, it must continue flight and complete landing on airfields initial or end points of route. The thrust-weight ratio of aircraft in this case must provide flight at the height/altitude not less than 4000-5000 m.

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Third condition for the selecting of thrust-weight ratio - from guarantee of standardized climb in the case of failure of one of engines - is examined in preceding/previous paragraph. According to formulas (9.9), (9.10) and (9.11) is determined the required thrust in the regime of the work of engines, which corresponds to the rate of climb and to calculated atmospheric conditions. Through the obtained required thrust (according to the appropriate engine characteristics) is found required total static thrust of engines.

Finally, are fourth condition, which can determine selection of thrust-weight ratio of passenger aircraft, demands of client for aircraft take-off roll during takeoff.

In this case

$$P_{\text{потр}} \cong 1{,}075 G_0 \left[\frac{1{,}172}{l_{\text{разб}} c_{y\,\max\,\text{взл}}} + \frac{1}{3}\left(\frac{1}{K'_{\text{взл}}} + 2 f_{\text{разб}}\right)\right] \text{kg}, \quad (9.14)$$

where $l_{\text{разб}}$ - prescribed/assigned takeoff run length m;

$c_{y\,\max\,\text{взл}}$ - maximum value c_y in takeoff position of high-lift device of wing;

$K'_{\text{взл}}$ - aerodynamic aircraft quality/fineness ratio with extended gear with high-lift device of wing in takeoff position and taking into account ground effect;

$f_{\text{разб}}$ - coefficient of friction with takeoff/run-up ($f_{\text{разб}}=0{,}03$—$0{,}05$).

From enumerated above four conditions is selected maximum value of required static thrust of engines or thrust-weight ratio of aircraft.

Number of engines is selected, on the basis of following contradicting each other conditions. From formulas (9.9), (9.10) and (9.11) it follows that the greater the number of engines, the less can be the thrust-weight ratio of aircraft and the less the operating costs; however, with the equal thrust-weight ratio aircraft with a smaller number of engines is more economical.

Solution of this problem taking into account both conditions makes it possible to select rational number of engines. Experiment of the solution of similar problems shows that:

- for the long-range main-line aircraft, which are operated, as a rule, on the airfields with the large lengths of runways (airfields of classes A and B) it is expedient to establish/install four engines (aircraft of Il-62, Boeing 707 and Boeing 747, Douglas DC-8, etc.);

- for main-line aircraft of medium range, which have some limitations on airfields (class C), optimum are installation of three engines (aircraft Tu-154), Boeing 727, Douglas DC-10, Lockheed L-1011, Hawker Siddeley "Trident", etc.);

- for main-line aircraft of short distance and for aircraft of local lines, intended for operation on airfields with short runways (class D) and even on unpaved airfields, is most advisable installation of two engines (aircraft Tu-134, Boeing-737, Douglas DC-9, BAC-111, etc.).

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Chapter XI.

SPECIAL FEATURES OF THE DESIGN OF AEROSPACE AIRCRAFT.

Continuous increase in space flights sharply raises the question of cost of delivery in space of payload by rocket systems. This leads to the search for the fundamentally new systems, which would make it possible to obtain the economically acceptable cost/value of flights.

Aerospace aircraft (VKS) is such system. Besides the delivery/procurement of people and loads from the Earth to the orbital stations and back, VKS will be necessary for service of scientific sapce laboratories, for the assembly of the interplanetary space vehicles or for their discharging after return to the earth's orbit and so forth [38].

In order to fulfill assigned missions, aerospace aircraft must satisfy following basic requirements.

1. VKS must be by repeatedly utilized flight vehicle.

2. VKS must derive/conclude payload in orbit with height/altitude of H=150-500 km.

3. VKS must possess good maneuverability in the atmosphere for liquidation of possible parallax of orbit (after start) and for accomplishing landing on assigned airfield (as usual aircraft).

4. VKS must possess sufficient maneuverability in space in order to complete orbital rendezvous and to carry out mating with assigned object.

Use of aerodynamic lift will make it possible to substantially decrease g-forces and to select trajectory of glide, acceptable on aerodynamic heating. Calculations show that even with hypersonic aerodynamic aircraft quality/fineness ratio $K_r=0.5-1$ deorbit it is possible to carry out with the g- force less than 2, in this case it will not be required the special orientation of crew relative to the vector of g- force and substantially will be reduced heat transfer rate in comparison with the ballistic entry.

Problem of guarantee of landing VKS in assigned place of Earth will be reduced to guarantee of necessary lateral distance in process of hypersonic gliding/planning, since guarantee of longitudinal distance will not apparently cause complication.

§ 1. Special features of the flight of aerospace aircraft.

For aerospace aircraft there is specific region of possible flights in the atmosphere and in space. Upper bound of flights in the atmosphere for VKS as winged of a flying apparatus is determined by the joint action of the force of gravity, aerodynamic force and centrifugal force, caused by the spherical surface of the Earth. Lower boundary of flights is determined by structural strength and by permissible temperature of aerodynamic heating (Fig. 11.1).

Lower boundary of the region of flights is general/common for all winged flying apparatuses. Upper boundary of the region depends on the special features of diagram (from the value of the specific wing load and the coefficient of aerodynamic lift).

To determine upper bound of flights VKS in the atmosphere is possible, examining conditions for level flight (gliding/planning) at given height/altitude.

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In the level flight, as is known, the weight of winged flight vehicle is balanced by two forces - aerodynamic lift and by centrifugal force, which appears as a result of moving the apparatus along curved path relative to the center of the Earth,

$$G = Y - P_{ц}. \qquad (11.1)$$

This equality will determine upper bound of flights VKS.

Expression for aerodynamic lift is common knowledge.

$$Y = c_y S \frac{\rho V^2}{2}.$$

Expression for centrifugal force in level flight can be registered thus:

$$P_{ц} = \frac{G (V_{г.п} \pm V_{в.з} \cos \varphi)^2}{g (R - H)}, \qquad (11.2)$$

where $V_{г.п}$ - speed of level flight;

$V_{в.з}$ - speed of rotation of Earth;

φ - angle of slope of plane of flight (orbit) to equatorial plane (orbit inclination);

H - flight altitude above surface of Earth (above sea level);

R - mean radius of Earth, R≈6370 km:

g - acceleration of gravity at height/altitude H

$$g = g_0 \frac{R^2}{(R+H)^2} = 9.81 \frac{R^2}{(R+H)^2}.$$

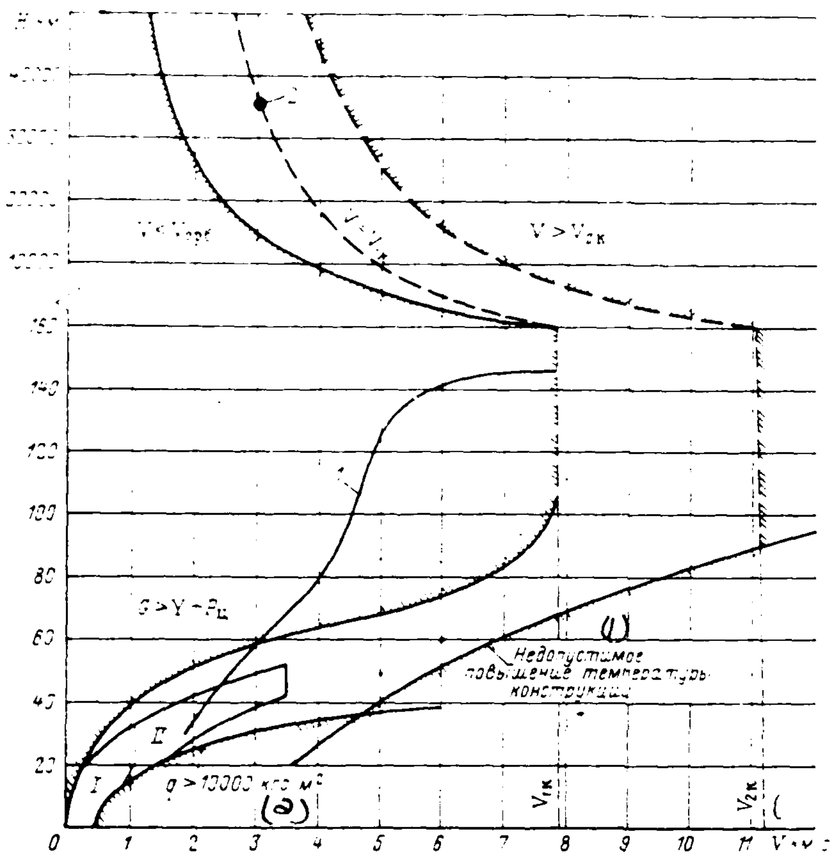

Fig. 11.1. region of possible flights VKS (I - region of the flights of contemporary aircraft; II - region of the flights of hypersonic aircraft): 1 - one of the possible ballistic trajectories of output/yield VKS into space (with the start from the hypersonic aircraft - carrier) 2 - synchronous orbit (revolving around the Earth along this orbit, flight vehicle will constantly remain above one point of equator) $V_{орб}$ - minimum speed, at which VKS can accomplish flight in space, moving along the circular or elliptic orbit; $V_{1к}$ - orbital velocity (circular); $V_{2к}$ - planet escape velocity (parabolic). Key: (1). Inadmissible increase of construction/design temperature.

(2). kgf/m². (3). km/s.

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For flight altitudes, where aerodynamic force still has vital importance (H<100 km), expression for centrifugal force with sufficient degree of accuracy it is possible to register thus:

$$P_{ц} = \frac{G(V_{г.п} - 460\cos\varphi)^2}{[illegible]}. \qquad (11.3)$$

Here $V_{[illegible]}$ in the m/s.

In flight of usual aircraft (V<1 km/s) centrifugal force can be disregarded/neglected (Fig. 11.2). On leaving into space it is necessary to consider not only centrifugal force, but also angle of orbit φ. If orbit considerably differs from polar, then launch flight vehicle more advantageous to the side of the daily rotation of the Earth. The possible angle of the orbit inclination to the equatorial plane with the start from any point of the Earth will be it is found in the range

$$\varphi_{ме} \leqslant \varphi \leqslant 90°,$$

where $\varphi_{мест}$ - local angle of the latitude (northern or southern) of launching point.

Solving equation (11.1) relative to value, ρ, let us determine value of mass density of air, and consequently, let us find necessary height/altitude, at which at given speed level flight

$$\rho = \frac{p}{c_y}\,\frac{62\cdot 10^6 - (V_{г.п} - 460\cos\varphi)^2}{3{,}1\cdot 10^7 V_{г.п}^2}, \qquad (11.4)$$

where p=G/S - specific load on lifting surface, is feasible.

Expression (11.4) determines so-called equilibrium height of flight.

Fig. 11.3 shows effect of speed (but for value $p/c_y=1000$ and angle of orbit inclination) to upper boundary of the region of flights VKS.

In flight in space aerospace aircraft becomes artificial Earth satellite. The motions of any celestial tolite (including artificial) are carried out, as is known, according to the laws of celestial mechanics, at basis of which lies/rests the law of universal gravitation of Newton. Therefore the region of the steady flights VKS in space will not have vital differences from a similar region of flights of contemporary artificial Earth satellites.

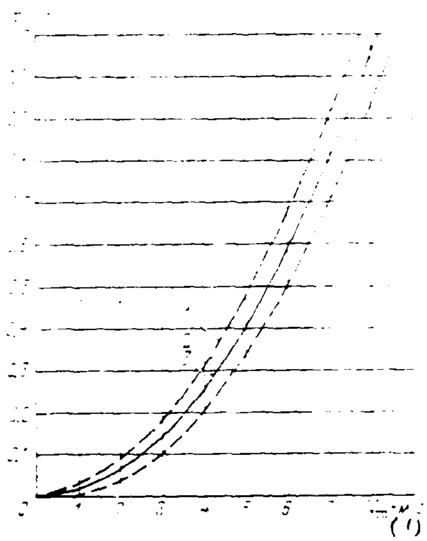

Fig. 11.2. Dependence of relation $P_\Sigma G$ on speed of level flight: 1 - flight in equatorial plane to side of daily rotation of Earth (φ=0); 2 - flight in plane of poles of Earth (φ=90°); 3 - flight in equatorial plane to opposite from rotation of Earth side (φ=0).
Key: (1). km/s.

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Flight trajectories VKS.

Motion of aerospace aircraft in general case is described by system of six differential equations, three of which reflect condition of equilibrium of forces in projections on axis of inertial coordinate system, and three - moment condition of equilibrium relative to these axes

$$m\left(\frac{dV_x}{dt}+V_z\omega_y-V_y\omega_z\right)=X;\quad m\left(\frac{dV_y}{dt}+V_x\omega_z-V_z\omega_x\right)=Y;$$

$$m\left(\frac{dV_z}{dt}+V_y\omega_x-V_x\omega_y\right)=Z;\quad J_x\frac{d\omega_x}{dt}+(J_z-J_y)\omega_y\omega_z=M_x;$$

$$J_y\frac{d\omega_y}{dt}+(J_x-J_z)\omega_z\omega_x=M_y;\quad J_z\frac{d\omega_z}{dt}+(J_y-J_x)\omega_x\omega_y=M_z,$$

where X, Y, and Z - projections of all external forces (including of reaction force) on appropriate coordinate axes;

M_x, M_y and M_z - moments of external and reaction forces relative to coordinate axes.

Since mass and moments of inertia VKS in the course of time are changed, that during solution of equations of motion it is necessary to accept

$$m=m(t);\quad J_x=J_x(t);\quad J_y=J_y(t);\quad J_z=J_z(t).$$

To solve system of equations of motion indicated is possible, if we represent in expanded/scanned form of expression of projections of external forces and moments/torques, entering right sides of equations.

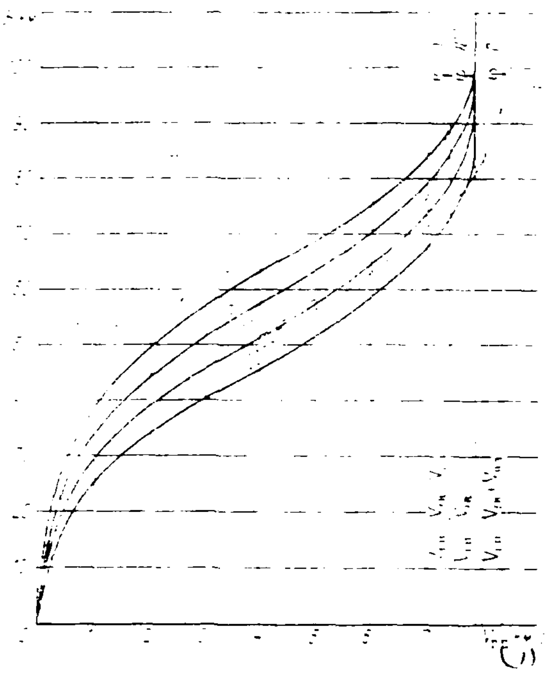

Fig. 11.3. Dependence of upper bound of flights VKS on speed.
Key: (1). km/s.

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On the flight vehicle the following external forces function:

- mass external forces, caused by the attraction of the Earth, Suns also of the Moon;

- aerodynamic forces (in flight in the sufficiently dense layers of the atmosphere);

- thrust of engine (with its work).

During detailed analysis of dynamics of flight VKS (for example, during navigational calculations) in resolving differential equations of motion it is necessary to consider all external forces, which function on flight vehicle. However, into the period of preliminary design VKS (during the selection of the diagram and basic parameters) it is possible to introduce the series/row of the assumptions, which will make it possible to considerably simplify the system of equations of motion. For example, if we do not examine the interplanetary flight of apparatus, then it is possible to be bounded to the account only of the mass attracting force of the Earth.

Considering flight VKS in vertical plane as point of variable mass, we disregard expenditure of fuel/propellant for balancing/trimming and for compensation for random moments of roll and yaw in process injection into orbit. In this case are considered the following forces: G, Y, X, P, moreover they accept the thrust of engine P in the general case that directed along its axis and that inclined toward the axis of aircraft (to the wing chord) at angle ψ (Fig. 11.4).

At high velocities of flight (see Fig. 11.2) it is necessary to also consider centrifugal force $P_{ц}$.

Besides enumerated forces, on flight vehicle will function Coriolis force, caused by daily rotation of Earth. With V≈3 km/s this force is approximately 0.02 G, while when $V=V_{1K}$ it reaches ~10% of gravitational force. The Coriolis force depends on the place of start and direction of flight and it must be considered during the navigational calculations. For the proximate analysis of motion VKS by the Coriolis forces it is possible to disregard.

Projecting/designing forces, which function on VKS, on axis of high-speed/high-velocity coordinate system and supplementing to obtained equations of motion kinematic constraints (connection/communication of change in altitude and flying range with speed and flight path angle), we will obtain necessary system of differential equations, which makes it possible to determine basic parameters of trajectory,:

$$\frac{dV}{dt}=\left[\frac{P\cos(\alpha-\varphi)-X}{G}-\sin\theta\right]g; \qquad (11.5)$$

$$\frac{d\theta}{dt}=\left[\frac{P\sin(\alpha+\varphi)+Y+\frac{GV^2\cos\theta}{g(R+H)}}{G}-\cos\theta\right]\frac{57.3g}{V}; \qquad (11.6)$$

$$dH/dt=V\sin\theta; \qquad (11.7)$$

$$dL/dt=V\cos\theta, \qquad (11.8)$$

where α - angle of attack;

θ - flight path angle to local horizon.

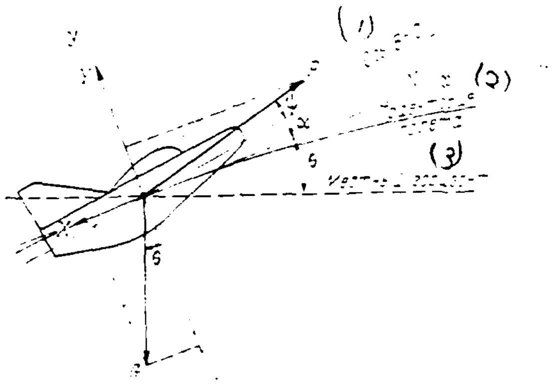

Fig. 11.4. Forces, which function on VKS in flight in vertical plane.
Key: (1). Axis. (2). trajectory of flight. (3). Local horizon.

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This system of equations can be solved by methods of numerical integration with use by computer(s) (during diploma design it is possible to use usual slide rule).

For definition of parameters of trajectory VKS at return from space entire phase of flight can be considered as equilibrium gliding/planning, in this case are valid following assumptions:

$$P=0;\quad \sin\theta\approx\theta=0;\quad \cos\theta=1;\quad G=\mathrm{const}.$$

Motion VKS in section of gliding/planning will be described by equations (11.1) and (11.5). The latter in this case will take the form

$$\frac{dV}{dt}=-\frac{X}{G}g. \tag{11.9}$$

From (11.9) it follows

$$t_{\text{пл}}=\frac{G}{g}\int\limits_{V_{\text{н. пл}}}^{V}\frac{dV}{-X}. \tag{11.10}$$

where $V_{н.пл}$ - speed at initial moment of gliding/planning (when $t_{пл}=0$).

Solving together equations (11.1) and (11.10), we will obtain expression for determining time of gliding/planning

$$t_{пл}=K_{г}\frac{R+H}{2V_{1к}}\ln\frac{(V_{н.пл}-V_{1к})(V_{к.пл}-V_{1к})}{(V_{н.пл}-V_{1к})(V_{к.пл}-V_{1к})}. \qquad (11.11)$$

where $K_{г}=c_y/c_x$ -hypersonic lift-drag ratio VKS;

$V_{1к}$ -orbital velocity;

$V_{к.пл}$ - speed at the end of gliding/planning.

In order to pass from orbital flight to regime of equilibrium gliding/planning, it is necessary to exert retro impulse, which ensures

$$\Delta V_{т}=30-70\ m/s.$$

The speed in the beginning of gliding/planning will be equal to

$$V_{н.пл}=V_{1к}-\Delta V_{т}.$$

Regime of equilibrium gliding/planning begins at altitude H=90-100 km (see Fig. 11.3). For the precomputations, assuming/setting R=6370 km; $V_{1к}=7850$ m/s; $V_{к.пл}\approx 0$, it is possible to determine the total time of gliding/planning on the approximate dependence, obtained from (11.11),

$$t_{п.пл}=2300\cdot K_{г}\ \text{s}. \qquad (11.12)$$

It must be noted that 75-80% of time gliding/planning occurs at speed V>5 km/s (Fig. 11.5).

Glide path always can be decomposed in individual sections with K_r=const. then gliding distance can be found from (11.9), having preliminarily multiplied left and right sides to V and after expressing G from (11.1):

$$\frac{1}{2}\frac{dV^2}{dt}=-\frac{g}{K_r}\left[1-\frac{V^2}{g(R+H)}\right], \qquad (11.13)$$

whence

$$L_{пл}=K_r\frac{R}{2}\ln\frac{V_{1к}^2-V_{к.пл}^2}{V_{1к}^2-V_{н.пл}^2}. \qquad (11.14)$$

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Equations (11.11) and (11.14) make it possible to determine $t_{пл}$ and $L_{пл}$ in any trajectory phase (i.e. for any values $V_{н.пл}$ and $V_{к.пл}$).

For precomputations gliding distance VKS from moment/torque of orbit ejection to touchdown can be determined from formula

$$L_{п.пл}=13\,800\cdot K_r \text{ км}. \qquad (11.15)$$

Let us note that ~90% of entire time of gliding/planning (on distance) occur at speed V>5 km/s (Fig. 11.6).

Distance of lateral maneuver depends on value $K_r^{1.5}$. In the precomputations complete distance of lateral maneuver can be determined according to the formula

$$L_{п.бок}=1400\cdot K_r^{1,5} \text{ км}. \qquad (11.15')$$

Problem regarding flight trajectory VKS in space coincides with task of determining orbits of celestial bodies (Kepler's task). The motion of body in this case is examined in the polar system coordinates with the pole in the center of Earth. The equations of motion of flight vehicle in the polar coordinate system can be obtained, projecting/designing the external forces, which function on the apparatus, for the direction of radius-vector and the tangent to the circle/circumference, described by radius-vector.

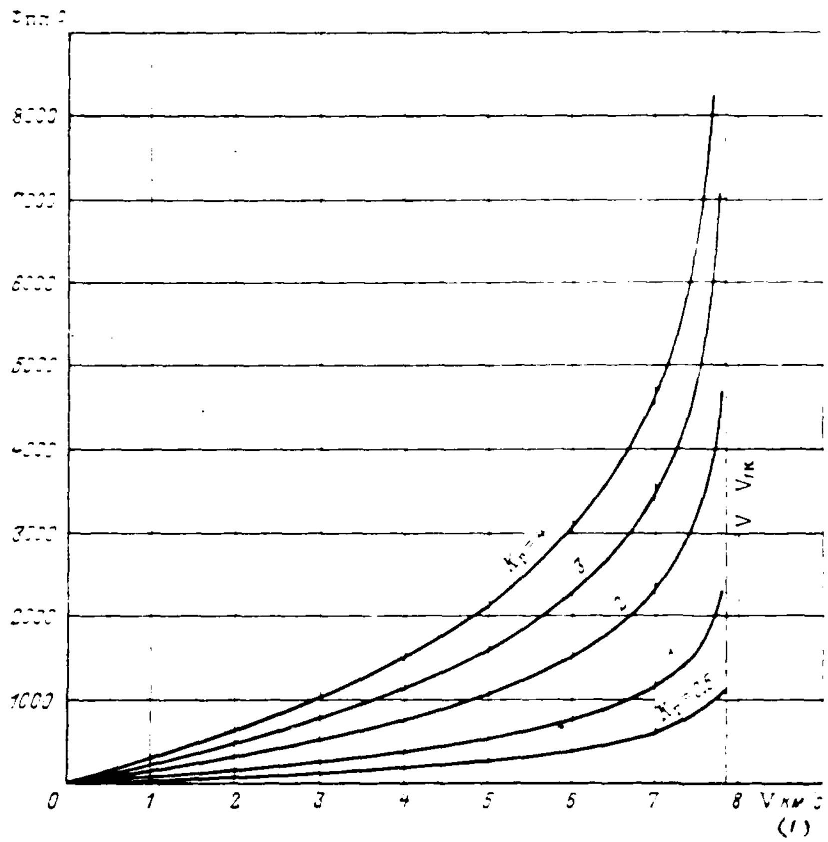

Fig. 11.5. Dependence of time of gliding/planning on speed and hypersonic lift-drag ratio VKS.

Key: (1). km/s.

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In particular, equations of motion VKS in space (in the absence of aerodynamic forces and thrust) will take the form

$$\frac{dV_S}{dt}+V_r\frac{d\varkappa}{dt}=0; \quad (11.16)$$

$$\frac{dV_r}{dt}-V_S\frac{d\varkappa}{dt}=-g_0\frac{R^2}{r^2}, \quad (11.17)$$

where V_S - peripheral component of velocity;

[illegible]-the radial component of speed;

κ - angle of rotation of radius-vector (vectorial angle), calculated off the polar axis, a certain initial unchangeable in the space direction of radius-vector;

r - distance from VKS to the center of the Earth (radius-vector).

Theory of motion of body under space conditions under action of forces of gravitation carry the name of elliptical theory. At present elliptical theory finds greatest use during the solution of such basic problems of cosmonautics as the determination of the orbits of artificial Earth satellites, the orbits of interplanetary flight vehicles, etc.

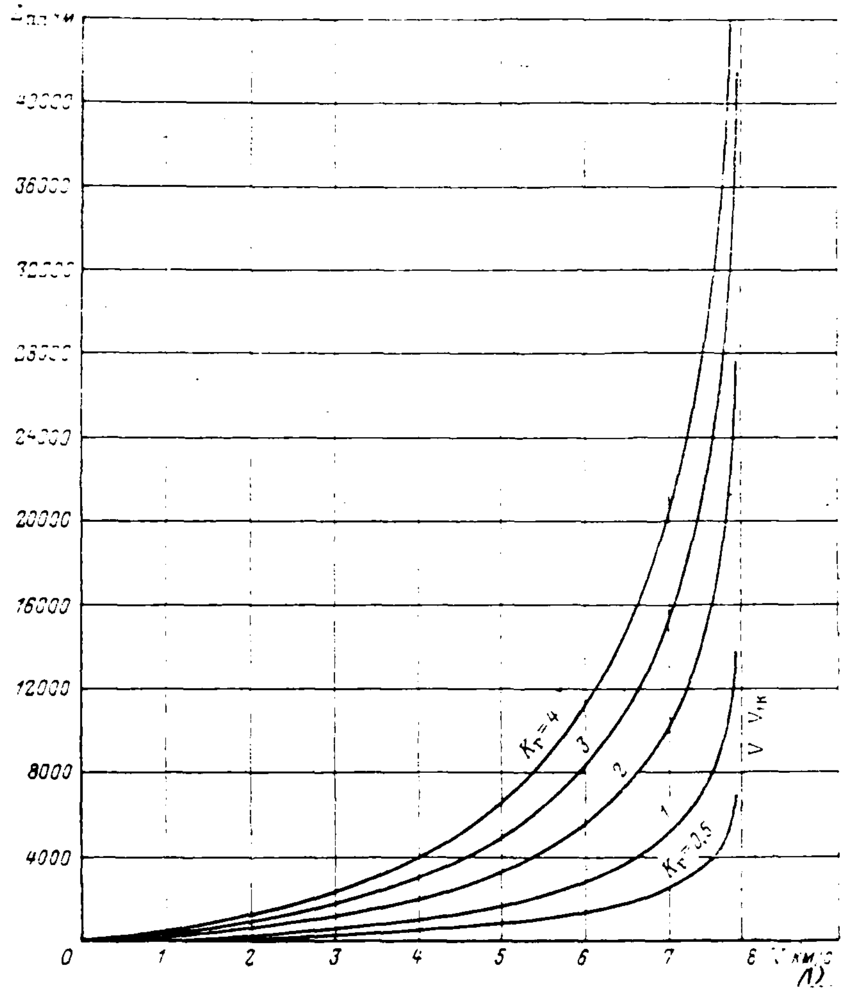

Fig. 11.6. Dependence of gliding distance on speed and hypersonic lift-drag ratio VKS.

Key: (1). km/s.

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This theory determines the flight trajectories VKS in space. Thus, the equation of orbit in the polar coordinates can be obtained, solving the system of differential equations (11.16) and (11.17):

$$r = \frac{p}{1 + e\cos(\varkappa - \varkappa_0)}, \qquad (11.18)$$

where p - focal orbital parameter;

e - orbit eccentricity;

$\varkappa_0$ - initial value of angle $\varkappa$.

During motion in terrestrial gravitational field, when focus of orbit located in center of Earth, value of focal parameter and eccentricity will be equal to

$$p=\frac{V_0^2 r_0^2 \cos^2\theta_0}{fM_3}; \quad (11.19)$$

$$e=\sqrt{1-\frac{2V_0^2 r_0 \cos^2\theta_0}{fM_3}+\frac{V_0^4 r_0^2 \cos^2\theta_0}{f^2 M_3^2}}. \quad (11.20)$$

Substituting these values in (11.18), we will obtain the final equation of orbit VKS (artificial Earth satellite)

$$r=\frac{\dfrac{V_0^2 r_0^2 \cos^2\theta_0}{fM_3}}{1+\sqrt{1-\dfrac{2V_0^2 r_0 \cos^2\theta_0}{fM_3}+\dfrac{V_0^4 r_0^2 \cos^2\theta_0}{f^2 M_3^2}}\cos(\varkappa-\varkappa_0)}. \quad (11.21)$$

where $r_0 R+H$ - initial distance from the center of Earth;

V_0 - the initial velocity in orbit (at height/altitude H from the surface of the Earth);

θ_0 - flight path angle to the local horizon at initial point;

fM_3 - constant of the gravitational field of the Earth;

f - gravitational constant;

M_3 - mass of the Earth.

According to law of universal gravitation weight of any body at height/altitude H from surface of Earth is equal to

$$G = f\frac{M_3 m}{(R+H)^2}, \qquad (11.22)$$

where m - mass of body.

The constant of gravitational field, consequently, will be equal to

$$fM_3 = g_0 R^2.$$

It is known that the form of the curve of second order is caused by the value of its eccentricity. With e=0 equation (11.18) is the equation of circle/circumference, when e<1 - equation of ellipse, when e=1 - equation of parabola and, finally, when e>1 - equation of hyperbola.

Equality θ_0=0 will be one of injection condition in orbit VKS; therefore in this case it is possible to consider that orbit eccentricity is determined by speed and height/altitude at initial point of orbit

$$e = e(V_0, H).$$

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Let us find the necessary initial velocity for the motion from the circular orbit. This speed carries name of circular, or first space (V_{1k}).

For case of e=0 from (11.20) we will obtain

$$V_0 = V_{1k} = \sqrt{\frac{g_0 R^2}{R+H}}. \qquad (11.23)$$

Circular orbit is special case. For its realization are necessary specified conditions ($V_0=V_K$ and $\theta_0=0$). Furthermore, as a result of the disturbances/perturbations, called mainly by the flattening of the form of the Earth, appear the deviations, which distort orbit shape. Therefore strictly circular orbit can be obtained only in the equatorial plane. However, during the determination of the parameters VKS it is possible to consider the form of the Earth sphere and to determine orbital velocity from formula (11.23).

For example, for altitude H=100 km numerical value of orbital velocity (at $\varphi=90°$)

$$V_K=7.85 \text{ km/s}.$$

With values of eccentricity $0<e<1$ equation (11.18) is equation of ellipse. Ellipse, as it is known, excentricity and focal parameter, is characterized by the an even major (a) and minor (b) semiaxis

$$a=\frac{p}{1-e^2};\quad b=a\sqrt{1-e^2}$$

Equations (11.19) and (11.20) make it possible to find major axis of elliptic orbit

$$2a=\frac{R-H}{1-\frac{V_0^2(R-H)}{2g_0R^2}}. \qquad 11.24$$

During motion to elliptic orbit flight altitude will continuously vary from minimum H (perigee) to maximum H+ΔH (apogee).

From expression (11.24) it is evident that when $V_0=V_K$ transverse will be equal to 2a=2(R+H), i.e., orbit is converted into circle/circumference.

At speed, equal to parabolic, or second space (V_{2k}), flight trajectory becomes parabola. The flight vehicle, which developed flight speed $V \geqslant V_{2k}$ to the earth is not returned.

Planet escape velocity is determined as follows:

$$V_{2k} = \sqrt{2} \cdot V_{1k} = \sqrt{\frac{2g_0 R^2}{R+H}}. \qquad (11.25)$$

For height/altitude of 100 km $V_{2k} = 11.1$ km/s.

Fig. 11.7 shows the possible orbits of space vehicle.

Time of one revolution VKS around Earth with circular orbit at height/altitude H is equal

$$t_{krug} = \frac{2\pi (R+H)}{V_{1k}} = \frac{2\pi}{\sqrt{g_0 R^2}} (R+H)^{3/2}. \qquad (11.26)$$

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For elliptic orbits orbital period is determined analogously, but instead of (R+H) into expression (11.26) it is necessary to substitute value of semimajor axis of ellipse

$$t_{el} = \frac{2\pi}{\sqrt{g_0 R^2}} a^{3/2}. \qquad (11.27)$$

After output/yield into space need for certain change in orbital parameters can emerge. To change the parameters of orbit (i.e. to pass from the circular orbit to the elliptical and back, or to change the angle of the orbit inclination) is possible, varying value and direction of flight speed.

If it is necessary, for example, to increase flight altitude by value ΔH, then it is necessary to communicate to flight vehicle further speed ΔV_H, equal to

$$\Delta V_H = V_0\left(\sqrt{\frac{R+H+\Delta H}{R+H+\Delta H/2}}-1\right). \qquad (11.28)$$

Orbit in this case will be elliptical and flight speed will vary from V_{max} at height/altitude H (perigee of orbit) to V_{min} at height/altitude H+ΔH (apogee of orbit).

During motion of body along orbit change in kinetic energy is equal to change in potential energy

$$d\left[\frac{mV^2}{2} - mg(R+h)\right]=0,$$

where h - spot height of orbit, speed of flight in which is equal to V. Since the mass of flight vehicle remains constant, then

$$\frac{V^2}{2} - \frac{g_0R^2}{R+h} = \mathrm{const.} \qquad (11.29)$$

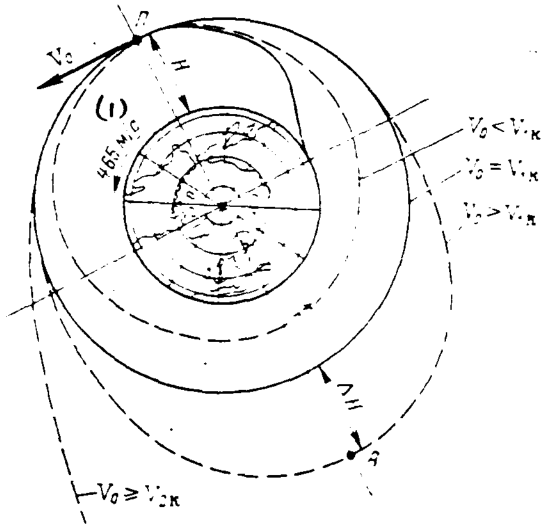

Fig. 11.7. Schematic of removal of flight vehicle in equatorial orbit (φ=0) during maximum use of daily rotation of Earth: П - perigee of orbit; A - apogee of orbit; П* - perigee of new orbit (when [illegible] and at sufficiently high value H).

Key: (1). m/s.

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This expression, called integral of energy, shows that flight speed will depend only on trajectory height at the particular point

From equation (11.29) we will obtain velocity in apogee of orbit (at height/altitude H+ΔH)

$$V_{min} = \sqrt{V_{max}^2 - \frac{2\Delta H g_0 R^2}{(R+H)(R+H+\Delta H)}}, \quad (11.30)$$

where V_{max} - speed in perigee of orbit.

Speed in perigee of orbit (i.e. new speed at height/altitude H), will be, obviously, is equal to $V_{max} = V_0 - \Delta V_H$.

At height/altitude (H+ΔH) occurs inequality $V_{min} < V_{cir}$, therefore, if it is necessary to increase flight altitude, after preserving in this case circular orbit, then flight speed at height/altitude (H+ΔH) must be increased to value V_{cir} at given height/altitude.

Simplest maneuver with respect to change in angle of orbit inclination to angle $\Delta\varphi$, without varying flight altitude, can be fulfilled by change in direction of flight speed to angle $\Delta\varphi$. For similar maneuver for flight vehicle it is necessary to furnish further speed ΔV_φ directed at angle $\left(90^\circ - \frac{\Delta\varphi}{2}\right)$ toward the plane of initial orbit (Fig. 11.8). The value of the further speed in this case will be equal to

$$\Delta V_\varphi = V_0 \frac{\sin \Delta\varphi}{\cos \frac{\Delta\varphi}{2}}. \qquad (11.3)$$

During design VKS possible change in speed for one or the other maneuver in space must be considered, since it requires further consumption of fuel sometimes of very essential.

Aerodynamic heating.

Flight at high temperatures is distinctive special feature of flight VKS in the atmosphere. The external sources of heating are: aerodynamic (kinetic) heating, solar radiation, radiation of the Earth and its atmosphere. Furthermore, are other heat sources, placed within the flight vehicle. The determining value for VKS will have aerodynamic heating. Other (external and internal) sources of heating in this case can be disregarded/neglected.

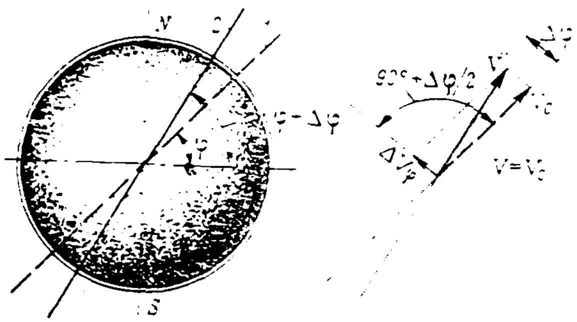

Fig. 11.8. Change in orbital plane with H=const: 1 - initial orbit; 2 - new orbit.

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Basic quantity of heat to skin/sheathing of apparatus is fed from boundary layer. Temperature of boundary-layer air of near to the temperature of stagnation. By considering air as perfect gas (with T<2000 K) and by taking into account heat exchange in a boundary layer, it is possible to find temperature on the internal boundary of the boundary layer:

a) laminar boundary layer

b) turbulent boundary layer

$$\left.\begin{aligned} T_{n.c} &\approx T_H(1+0.17\,M^2); \\ T_{n.c} &\approx T_H(1+0.18\,M^2). \end{aligned}\right\} \quad (11.32)$$

Here T_H - temperature of air at height/altitude H.

Determination of temperature of skin heating. Under the conditions, when heat exchange is determined by the combined action of convection, thermal conductivity and emission, the temperature of skin/sheathing of flight vehicle can be determined from the equation of the balance of heat:

$$q_{п.с}-q_{изл}=c\gamma\delta\cdot\frac{dT_{об}}{dt}. \qquad (11.33)$$

where $q_{п.с}$ - heat transfer rate from the boundary layer, i.e. a quantity of heat, which enters skin element of area per unit time in kkal/m²s;

$q_{изл}$ - heat transfer rate, by the emitted skin/sheathing into the surrounding space;

c - specific heat of skin material in kkal/kgs×grad;

γ - specific weight/gravity material of skin and kG/m³;

δ - thickness of skin/sheathing m;

t - time in s;

$T_{об}$ - to temperature of the external surface of skin/sheathing in deg K.

Equation (11.33) describes unsteady thermal process, during which temperature of body surface is changed in the course of time. The solution of this nonlinear differential equation can be obtained by the methods of numerical integration.

Greatest temperature of skin/sheathing will be when $dT_{об}/dt=0$. In this case occurs the steady heat exchange and the equilibrium temperature of skin/sheathing, which is set during the endurance flight under the constant/invariable conditions. In this case $q_{п.с}=q_{изл}$.

Emitted by skin heat transfer rate is determined according to the law of Stephan - Boltzmann

$$q_{изл}=\varepsilon\sigma T_{об}^{4}. \qquad (11.34)$$

where ϵ - radiation coefficient, or degree of dark of skin/sheathing;

σ=1.37·10⁻¹¹ kkal/m²s·grad⁴ - radiation coefficient of blackbody.

Coefficient ϵ considers radiating capacity of body (skin/sheathing) in comparison with blackbody. It depends on the material of surface and its treatment, and also on temperature. For skin VKS it is possible to accept $\epsilon \approx 0.8$.

Heat transfer rate, which enters skin/sheathing from boundary layer, in accordance with Newton's law, is determined as follows:

$$q_{\text{п.с}} = \alpha (T_{\text{п.с}} - T_{\text{сб}}), \qquad (11.35)$$

where α - local coefficient of convection heat transfer on boundary air - skin/sheathing in kkal/ms·grad.

Taking into account equality $q_{\text{изл}}$ and $q_{\text{п.с}}$, we will obtain equation, which makes it possible to determine temperature of skin/sheathing during steady heat exchange,

$$\epsilon \sigma T_{\text{об}}^4 + \alpha T_{\text{сб}} - \alpha T_{\text{п.с}} = 0. \qquad (11.36)$$

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Coefficient of convection heat transfer α has different values for plate and for critical point of spherical body. The approximation of the coefficient of heat transfer for the plate takes the form:

$$\alpha \approx 0.5 g \rho V c_p c_f \mathrm{Pr}^{-\prime \bullet}. \qquad (11.37)$$

where c_p - heat capacity of the air at a constant pressure in kkal/kgs•;GRad;

c_f - coefficient of air friction against the surface of skin/sheathing, depending on a number of Reynolds and structure of boundary layer;

$\mathrm{Pr} = \mu c_p g/\lambda$ - Prandtl number;

μ - coefficient of the viscosity of air in kg•s/m²;

λ - coefficient of the thermal conductivity of air in kkal/m•s×grad.

The physical constants of air c_p, μ, λ, which depend on the temperature, must be taken for the so- called determining temperature

$$T_{on} = T_H + 0{,}72\,(T_{n.c} - T_H). \qquad (11.38)$$

Prandtl number depends on the temperature of air (table 11.1). At large temperatures ($T_{on} > 1250$ K) Prandtl number can be considered constant.

For approximate estimate of temperature of skin/sheathing coefficient α can be determined thus: laminar boundary layer

$$\alpha = 31{,}6\, g \rho V c_p\, (\rho V x/\mu)^{-[illegible]}\, \mathrm{Pr}^{-[illegible]}; \qquad (11.39)$$

turbulent boundary layer

$$\alpha = 0{,}184\, g \rho V c_p\, (\rho V x/\mu)^{-[illegible]}\, \mathrm{Pr}^{-[illegible]}, \qquad (11.40)$$

where

$$x \approx \frac{0{,}0038}{\left(1 - \frac{120}{T_H}\right) \rho M}. \qquad (11.41)$$

Given formulas are valid for determining temperature of wing skin, tail assembly and cylindrical part of fuselage (housing).

For coefficient of convection heat transfer near critical point with laminar boundary layer it is possible to take following expression:

$$\alpha=0.54\,\mathrm{Pr}^{-0,66}gc_p\sqrt{\beta\rho\mu/r}, \qquad (11.42)$$

where r - radius of nose section of body;

$$\beta=2.82\sqrt{\frac{p}{p_c}\,\frac{1}{(1+0.2M^2)^{2.5}}}. \qquad (11.43)$$

Here p and p_c - pressure of the flow before and after the normal shock of pressure. Relations p/p_c are determined according to the known formula of gas dynamics

$$\frac{p}{p_c}=11.3\,\frac{(1+0.2M^2)^{3,5}}{1,167M^2-0,167}\left[\frac{2,43(1-0.2M^2)-2,5}{M^2}\right]^{3,5}.$$

Using equation (11.36) and formulas given above, it is possible to calculate equilibrium temperature of skin/sheathing VKS (and also any other aircraft) during aerodynamic heating.

Table 11.1.

$T_{\text{оп}}$ K	500	750	1000	1250
Pr	0,69	0,665	0,655	0,65

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Fig. 11.9 gives approximate values of equilibrium temperature for probable region of flights VKS (for small angles of attack).

Intensity of aerodynamic heating of surface substantially is reduced with increase in distance from leading wing edge (from nose of fuselage) and with increase in sweep angle of wing (tail assembly). Fig. 11.10 shows isotherms on the surface of hypersonic aircraft in the cruise at the height/altitude of 34 km with M=8. Should be focused attention on the temperature distribution along the lower intake plane (temperature distribution at the plane at the constant angle of attack and the zero angle of sweepback).

§ 2. Selection of the schematic of aerospace aircraft.

If we by payload understand weight of apparatus, concluded in orbit (without considering, naturally, weight of structure of latter/last step/stage of accelerator, which will also reach orbital speed), then by most adequate/approaching criterion for analysis and selection of diagram will be relative of useful payload

$$\overline{G}_{\text{п.в}}=G_{\text{п.в}}G_0.$$

Where $G_{\text{п.в}}$ - weight of vehicle, injected into orbit;

G_0 - launch weight of system.

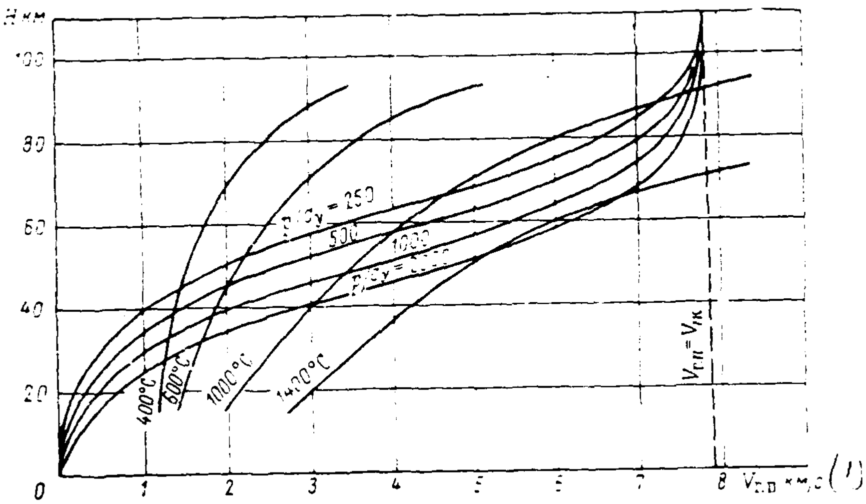

Fig. 11.9. Balanced temperature of flat surface at a distance of x=1.5 m from leading edge (ϵ=0.8).

Key: (1). km/s.

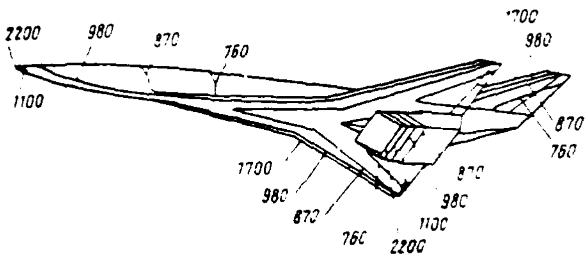

Fig. 11.10. Steady temperature in surface of aircraft (v °C) during endurance flight (V=2400 m/s H=34 km).

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Although this criterion is not comprehensive (more common criterion it is cost/value of system), nevertheless it plays main role, since on $G_{\text{п.н}}$ cost of delivery 1 kg of payload in orbit depends, other conditions being equal.

With given value of payload weight criterion of evaluation of diagram VKS will be, obviously, value of launching weight of system.

Possible schematics of aerospace aircraft. The most important characteristic of flying apparatuses for the delivery/procurement of load in orbit is the minimally necessary flight speed in orbit V_{op} (for flight altitude H<500 km $V_{op} \approx V_{k}$). A strict guarantee of the assigned flight speed not for one flight vehicle has this important value as for the space vehicles. For example, if aircraft was projected/designed for the flight speed V=3185 km/h (M=3), and in actuality speed proved to be to 1% less, i.e. V=3153 km/h (M=2.97), then this virtually in no way will be reflected in the efficiency of this aircraft. For the orbital apparatus the error in speed to 1% (i.e. instead of 7.8 km/s to obtain 7.72 km/s) indicates starting/launching idle, since apparatus will not be held in orbit and it will complete landing, having fulfilled not one turn around the Earth. Therefore reaching/achievement of the corresponding flight speed is the most important and necessary flight condition along the orbit. This condition will to a considerable degree determine the diagram and the basic parameters of orbital flight vehicles.

To find connection/communication of orbital speed with basic parameters of flight vehicle is possible, analyzing process of dispersal/acceleration and climb on leaving in orbit.

In process of dispersal/acceleration and climb flight vehicle acquires speed, which in general case can be registered thus:

$$V = V_{и} - \Delta V_{п} - V_{гр} \qquad (1.44)$$

where $V_{и}$ - ideal velocity of apparatus, i.e., speed which apparatus would obtain in the absence of force of gravity and force of aerodynamic resistance;

$\Delta V_{п}$- total losses of velocity from action of gravitation and aerodynamic drag;

$V_{ст}$ - starting speed of apparatus (for single-stage apparatuses with start from Earth, obviously, $V_{ст}=0$).

Ideal velocity of apparatus is determined from known formula K. E. Tsiolkovskiy

$$V_{и}=W_{e}\ln m_{нач}/m_{кон},$$

where W_{e} -effective exhaust velocity;

$m_{нач}$ - the initial mass of apparatus;

$m_{кон}$ - the finite mass of apparatus (after burnout).

Expressing effective discharge velocity through specific impulse (specific thrust of engine), and mass of flight vehicle through weight, we will obtain

$$V_{и}=g_{0}J_{т}\ln\frac{G_{нач}}{G_{кон}}=9.81J_{т}\ln\frac{1}{1-\bar{G}_{т}}, \qquad (11.45)$$

where

$J_{т}$-specific jet firing (to irrigation); $\bar{G}_{т}=G_{т}/G_{нач}$ - over-all payload ratio of fuel/propellant.

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On value $\Delta V_{п}$ velocity of start has especially essential effect.

For concrete/specific schematic of flight vehicle on leaving in orbit along specific trajectory value $\Delta V_{\text{п}}$ can be determined, integrating equations of motion by (11.5)- (11.7).

In period of preliminary design this value can be taken approximately; with sufficient degree of accuracy it is possible to remove/take it from graph Fig. 11.11, which is obtained as a result of trajectory calculation of injection into orbit with height/altitude of H=120-200 km; in this case different schematics of flight vehicles with actually possible parameters were examined.

Injection into orbit with H>200 km it is possible to decompose into two stages:

1) output/yield to H≈150 km;

2) maneuver on increase in altitude of orbit by values ΔH [see (11.28)].

Required quantity of fuel/propellant $G_{\text{т.пот}}$ for injection into orbit we will obtain, accepting in (11.44) $V=V_{1\text{к}}$ and deciding together (11.44) and (11.45) relatively $\overline{G}_{\text{т}}$.

$$\ln\frac{1}{1-\overline{G}_{\text{т.пот}}}=\frac{V_{1\text{к}}+\Delta V_{\text{п}}-V_{\text{ст}}}{9{,}81J_{\text{т}}},$$

or

$$\overline{G}_{\text{т.пот}}=1-\frac{1}{e^{\frac{V_{1\text{к}}+\Delta V_{\text{п}}-V_{\text{ст}}}{9{,}81J_{\text{т}}}}}, \qquad (11.46)$$

where $\overline{G}_{\text{т.пот}}=G_{\text{т.пот}}/G_{\text{нач}}$;

$G_{\text{нач}}$ -initial weight of apparatus (when $V=V_{\text{ст}}$), for single-stage apparatuses $V_{\text{ст}}=0$ and $G_{\text{нач}}=G_0$.

Fig. 11.12 depicts graphic interpretation of equation (11.46) for several values of starting speed.

Possible values $\bar{G}_T$ and $\bar{G}_K$ for contemporary aircraft and aircraft of nearest future are given in Fig. 11.13.

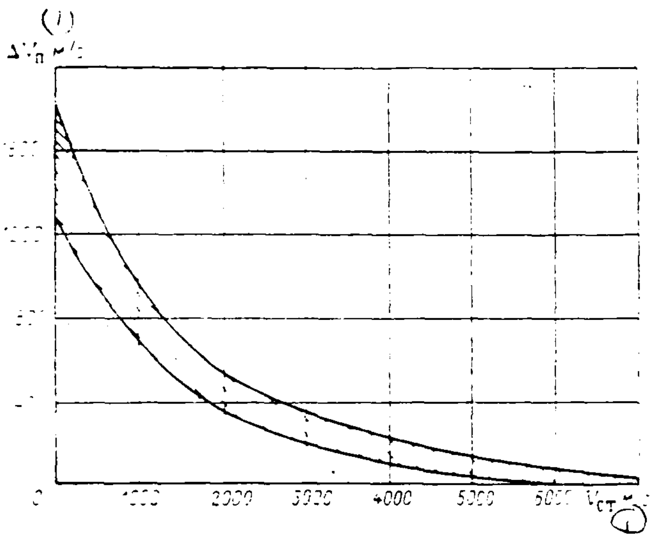

Fig. 11.11. Effect of speed of start on total speed losses from action of gravitation and aerodynamic drag (for orbits H=120-200 km). Key: (1). m/s.

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For injection into orbit flight vehicle with ZhRD when $J_{T} = 250-450$ must have over-all payload ratio of fuel/propellant $\bar{G}_{T} = 0.98 \div 0.87$ when $M_{CT} = 0$ (see Fig. 11.12).

If flight vehicle will start from hypersonic aircraft of carrier (for example, when $M_{CT} = 6-12$), then necessary fuel reserve for injection into orbit decreases; however, in this case it will be equal to $\bar{G}_{T.\text{ПОТ}} = 0.78-0.64$ for $J_{T} = 450$ (fuel/propellant H_2+O_2).

Only start with $M_{CT} \geqslant 18$ gives possibility of injection into orbit ($\bar{G}_{T.\text{ПОТ}} \leqslant 0.45$).

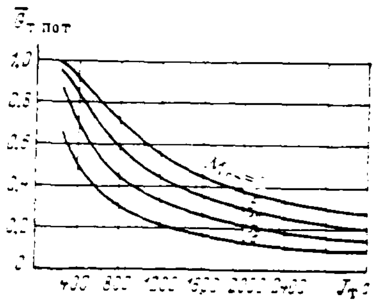

Fig. 11.12.

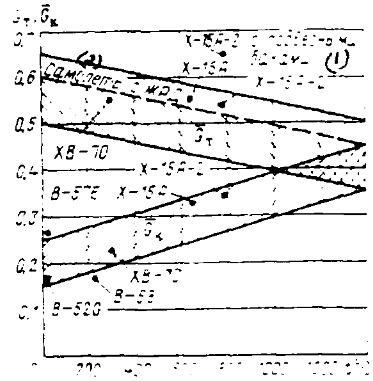

Fig. 11.13.

Fig. 11.12. Dependence of over-all payload ratio of fuel/propellant, required for injection into orbit (H=120-200 km), on specific jet firing.

Fig. 11.13. Change in over-all payload ratio of fuel/propellant and design of aircraft in dependence on operating temperature of design.

Key: (1). with suspended tanks. (2). aircraft with ZhRD.

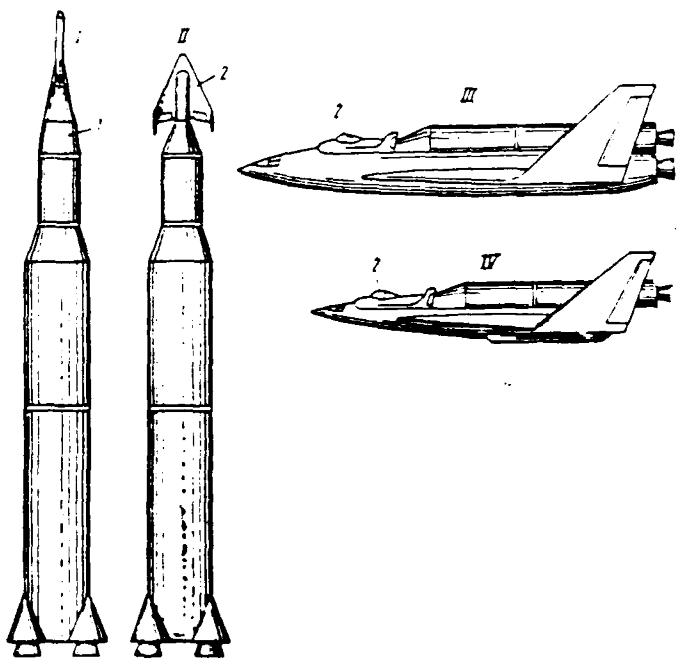

Fig. 11.14. Possible schematics of flight vehicles for delivery/procurement of payload in orbit (as fuel for all engines it is used liquid hydrogen): 1 - spacecraft with ballistic entry into the atmosphere; 2 - VKS.

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Therefore it is possible to do injection relative to schematic of apparatus: to deliver payload into orbit and to complete return and landing to the earth at present (when $J_{T} \approx 450$ s) can only multistage flight vehicles. Fig. 11.14 shows the possible schematics of such flight vehicles.

Space vehicle with ballistic entry into the atmosphere is payload in diagram I; landing to the earth is produced with use of parachutes. The apparatuses of diagrams II, III, IV have identical payload - the aerospace aircraft, which accomplishes planning/gliding entry into the atmosphere and horizontal landing to the earth.

Diagrams I and II are identical and are characterized by only payload. These diagrams are based on the principle of the maximum use of the existing constructions/designs of rockets. The high cost/value of carrier rocket leads to the need for the searches for the repeated use of steps/stages. The rescue of first stage is most expedient, since in this case from 75 to 80% of weight of the entire structure of carrier rocket it is returned conversely.

Hypersonic carrier aircraft (booster), piloted by crew and which is independently returned to place of start after starting/launching of space vehicle (diagram III and IV), gives best possibilities in creation of repeatedly utilized space systems. Upon transfer from the ballistic to the winged aerospace systems of repeated application the cost of the delivery of payload in orbit considerably will be reduced.

Diagram III is one of possible versions of rescue of first step of carrier rocket with ZhRD.

Diagram IV is most promising. during the use in the process of dispersing/accelerating the high specific impulse VRD very large gain in useful load, concluded in orbit, can be obtained in comparison with the carriers, equipped with ZhRD. Carrier with VRD will be close to the aircraft according to diagram and accomplishing of operations.

Efficiency of carrier with VRD is visible from Fig. 11.15.

Certain representation about weight distribution of apparatuses of diagrams examined gives Fig. 11.16.

One should note that diagram IV ensures not only repeated use of expensive first step/stage, it makes it possible to increase substantially values $\bar{G}_{\text{п.н.}}$ For the injection into orbit of one and the same payload the launching weight of the flight vehicle, planned according to diagram IV, will be two times less in comparison with the best samples of the contemporary carrier rockets:

$\bar{G}_{\text{п.н}}=0.10-0.12$ (diagram IV);

$\bar{G}_{\text{п.н}}=0.05-0.06$ (diagram I, II).

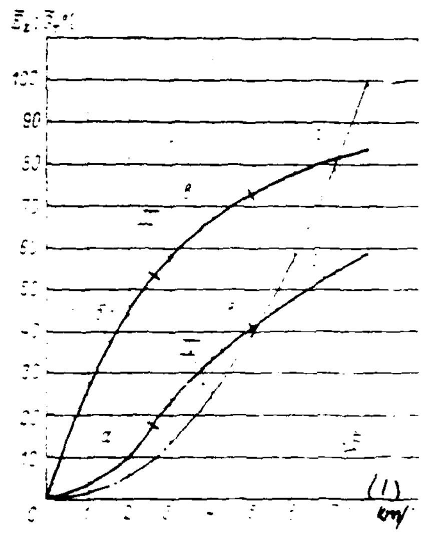

Fig. 11.15. Dependence of relative energy, to communicated payload on leaving in orbit, and relative fuel consumption from speed for diagrams III and IV in Fig. 11.14 (as fuel for all engines it is used liquid hydrogen: a) carrier aircraft with VRD: b) first stage with ZhRD: c) second step/stage with ZhRD: d) third step/stage with ZhRD:

Key: (1). km/s.

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During use of nuclear rocket engines (YaRD), whose specific impulse will considerably exceed specific impulse of ZhRD, diagram IV will be changed in direction of decrease of quantity of steps/stages.

Thus, when $J_T \approx 1000$ with VKS with YaRD, starting from carrier aircraft, it will leave in orbit without further accelerators. When $J_T \approx 2000$ s will drop out the necessity also for the carrier aircraft, since VKS in this case according to equation (11.46) can independently (starting from the Earth) leave in orbit.

During wide use of YaRD with specific impulse ~2000 s era of rockets as flight vehicles for injection of payload into orbit, apparently, will end, since key advantage of rocket steps/stages - high load ratio on fuel/propellant - it will lose its value, since required fuel load for injection into orbit when $J_T > 2000$ will be $\bar{G}_{T\,\text{пот}} < 0.4$ (see Fig. 11.12).

Schematic of aerospace aircraft must provide:

- obtaining necessary value of lift-drag ratio in hypersonic and subsonic flight conditions;

- light thermal loads upon entry into the atmosphere.

Apparatus with $K_r = 0$ is tested/experienced upon entry into the atmosphere g-force from 8 to 10. An increase in the lift-drag ratio in all to 0.5 makes it possible to decrease the g-force in the entire line of descent up to two.

Research showed that in majority of cases hypersonic lift-drag ratio of VKS can be bounded by value

$$K_r = 1 - 2.$$

Fig. 11.17 shows change in physical characteristics of VKS in dependence on value K_r. A gain in weight of apparatus with an increase in the hypersonic lift-drag ratio is connected with an increase in the ratio of surface area to working volume, with an increase in the duration of flight, which leads to greater total thermal loads.

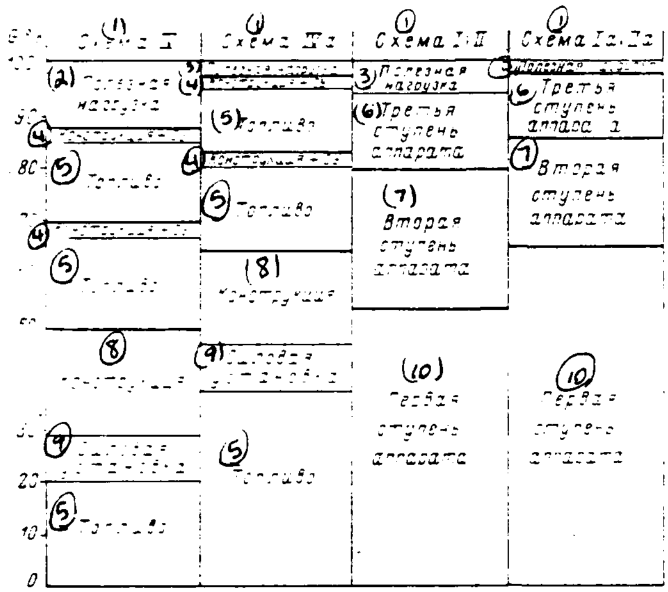

Fig. 11.16. Diagram of weight distribution of flight vehicles for delivery/procurement of payload in orbit (with start from Earth); in diagrams all I, II, IV engines they work on liquid hydrogen, in diagrams Ia, IIa, IVa all engines work on kerosene.

Key: (1). Diagram. (2). Payload. (3). Payload. (4). Construction/design + SU. (5). Fuel/propellant. (6). Third step/stage of apparatus. (7). Second step/stage of apparatus. (8). Construction/design. (9). Power plant. (10). First stage of apparatus.

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Acceptable landing data of VKS are provided with value of subsonic lift-drag ratio not less than four.

Widespread investigations of aerodynamic shapes of maneuverable aerospace apparatuses for repeated application at present are conducted, in this case special attention is paid to apparatuses with lifting body. The less complicated resolution of the problem of the thermal insulation of construction/design is the major advantage of such apparatuses (in the comparison with the winged). For an improvement in the subsonic and landing data of VKS with the lifting body it is proposed to use a special wing with the subsonic profile/airfoil (with M>1 wing it is removed).

Evolution of diagram of VKS is shown in Fig. 11.18.

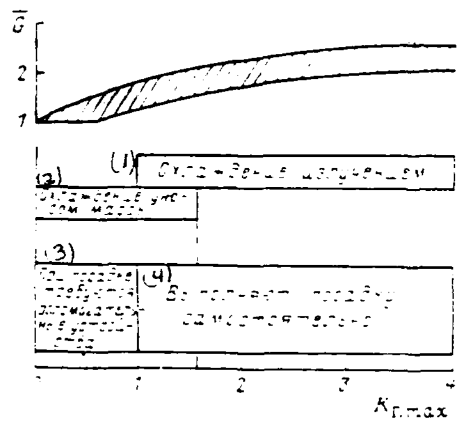

Fig. 11.17. Effect of hypersonic lift-drag ratio on characteristics of VKS.

Key: (1). Radiation cooling. (2). Cooling by ablation. (3). During landing auxiliary devices/equipment are required. (4). Landing is implemented independently.

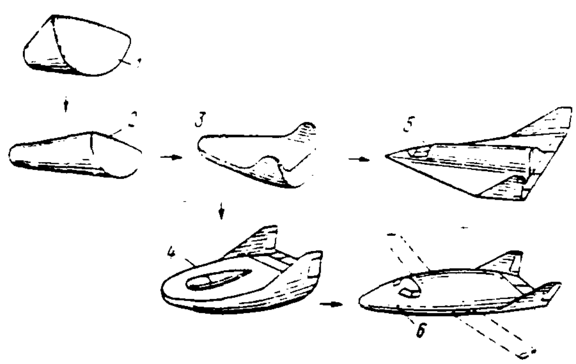

Fig. 11.18. Evolution of diagram VKS: 1 - blunted semicone with aperture angle of 60°; 2 - blunted semicone with aperture angle of 30°; 3 - semicone with endplates; 4 - VKS with lifting body (lower surface oval); 5 - winged VKS; 6 - VKS with lifting body (lower surface flat/plane).

(1) Схема	M > 10	M < 1	(1) Схема	M > 10	M < 1
1	$K_{-} = 0.5$	$K = 0.8$	4	$K_r = 1.3$	$K = 4$
2	$K_r = 1.2$	$K = 2$	5	$K_r = 2$	$K = 8$
3	$K_r = 1.2$	$K = 4$	6	$K_r = 2$	$K = 8$

Key: (1). Diagram.

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§ 3. Determination of the basic parameters of aerospace flight vehicle.

Let us consider basic parameters for panoramic sketches.

We will consider that optimum parameters correspond $(\overline{G}_{п.н})_{max}$ in dispersal/acceleration to $V_{к}$.

Fig. 11.19 shows project of VKS, designed for 10-12 people.

Apparatus of multistage circuit.

Schematic of apparatus in general case they will determine: carrier aircraft (retained step/stage), accelerators (nonretained steps/stages) and VKS (retained step/stage) (see Fig. 11.14, diagram IV).

Launching weight of apparatus in this case is equal to

$$G = G_{н} - G_{1} - \ldots + G_{m} - G_{в.с}. \qquad (11.47)$$

where $G_{c.н}$ - weight of carrier aircraft; G_1 - weight of first accelerator; G_m - weight of m accelerator; $G_{ВКС}$ - weight of VKS; m - number of accelerators.

For determining optimum values of parameters of similar flight vehicle it is necessary to answer two questions:

1) what payload and to what speed (in given parameters) it is capable to drive away carrier aircraft;

2) what part of this load it is capable of leaving on near-earth orbit. In other words, it is necessary, providing the maximum of criterion $\bar{G}_{п.н}=G_{ВКС}/G_0$, to find minimum launching weight of system, if concrete/specific weight of VKS is prescribed/assigned or, if is intended concrete/specific carrier aircraft, to find a maximally possible weight of VKS, capable of leaving in orbit with the start from this carrier aircraft.

Let us introduce following concept and designations: G_i - weight of i step/stage; $G_{н i}$ - useful load, accelerated/dispersed with i step/stage to speed $V=V_{ст i}+\Delta V_i$ and height/altitude $H=H_{ст i}+\Delta H_i(V_{ст i})$ and $H_{ст i}$ - respectively speed and height/altitude of start of i step/stage; ΔV_i and ΔH_i - further speed and height/altitude, of acquired by load $G_{н i}$ due to fuel/propellant i steps/stages); $G_{т i}$ - fuel load of i step/stage, required for dispersing/accelerating load with weight $G_{н i}$ (to value ΔV_i and ΔH_i):

$$\bar{G}_{т i}=\frac{G_{т i}}{G_i-G_{н i}}; \quad \bar{G}^{*}_{т i}=\frac{G_{т i}}{G_i}.$$

For the carrier aircraft we will have:

$$G_{\text{н.с.н}} = G_{\text{у}1} + \ldots + G_{\text{у}m} + G_{\text{внс}};$$

$$\overline{G}_{\text{т.с.н}} = \frac{G_{\text{т.с.н}}}{G_{\text{с.н}} - G_{\text{н.с.н}}} = \frac{G_{\text{т.с.н}}}{G_0}; \quad \overline{G}^{\bullet}_{\text{т.с.н}} = \frac{G_{\text{т.с.н}}}{G_{\text{с.н}}}.$$

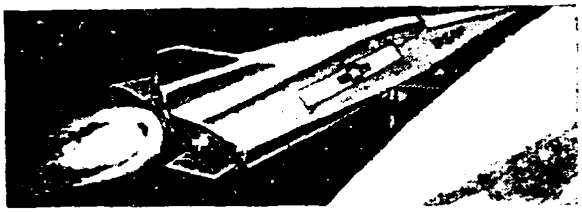

Fig. 11.19. VKS of firm "Lockheed", USA (drawing).

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For first accelerator:

$$G_{\text{н.}y1} = G_{y2} + \ldots + G_{ym} + G_{\text{ВКС}};$$
$$\overline{G}_{\text{т.}y1} = \frac{G_{\text{т}y1}}{G_{y1} - G_{\text{н.}y1}}; \quad \overline{G}'_{\text{т}y1} = \frac{G_{\text{т}y1}}{G_{y1}}.$$

For m accelerator:

$$G_{\text{н.}ym} = G_{\text{ВКС}}; \quad \overline{G}_{\text{т.}ym} = \frac{G_{\text{т.}ym}}{G_{ym} - G_{\text{ВКС}}}; \quad \overline{G}'_{\text{т}ym} = \frac{G_{\text{т}ym}}{G_{ym}}.$$

Further speed, communicated by i step/stage to accelerated/dispersed load, in general case will be equal to

$$\Delta V_i = V_{\text{и}i} - \Delta V_{\text{п}i},$$

where $V_{\text{и}i}$ - ideal velocity, which communicates i grade of load $G_{\text{н}i}$;

$\Delta V_{\text{п}i}$ - speed losses from action of gravitation and aerodynamic drag in acceleration phase from $V_{\text{ст}i}$ to $V_{\text{ст}i+1}$.

It is easy to show that

$$\overline{G}_{\text{т}i} = \frac{G_{\text{т}i}}{G_i - G_{\text{н}i}} = \overline{G}'_{\text{т}i}\left(1 - \frac{G_{\text{н}i}}{G_i - G_{\text{н}i}}\right),$$

and consequently, according to equation (11.46), will occur equality

$$\overline{G}^{*}_{\tau i}\left(1-\frac{G_{\text{н}i}}{G_i-G_{\text{н}i}}\right)=1-\frac{1}{e^{\frac{\Delta V_i-\Delta V_{\text{п}i}}{9{,}81 I_{\tau i}}}}.$$

From this equality let us find weight of apparatus when $V_{\tau i}$ at moment of firing engines of i step/stage (after department/separation from i step/stage),

$$G_i-G_{\text{н}i}=\frac{\overline{G}^{*}_{\tau i}}{\overline{G}^{*}_{\tau i}-1+\frac{1}{e^{\frac{\Delta V_i-\Delta V_{\text{п}i}}{9{,}81 I_{\tau i}}}}}G_{\text{н}i}. \qquad (11.48)$$

Since for each previous step/stage accelerated/dispersed load is sum of all subsequent steps/stages, then on the basis of dependence (11.48) launching weight of multistage flight vehicle will be equal to

$$G_0=\left(\prod_{i=1}^{n-1}\frac{\overline{G}^{*}_{\tau i}}{\overline{G}^{*}_{\tau i}-1+\frac{1}{e^{\frac{\Delta V_i-\Delta V_{\text{п}i}}{9{,}81 I_{\tau i}}}}}\right)G_{\text{п.н}}. \qquad (11.49)$$

where n - number of steps/stages of apparatus; sign $\prod_{i=1}^{n-1}$ - indicates product (n-1) of terms; n-1=m+1 - number of "working" steps/stages of apparatus (fuel/propellant of which is expended/consumed in process of dispersal/acceleration); $G_{\text{п.н}}$ - payload weight, concluded in orbit.

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As was accepted, $G_{\text{пн}}=G_{\text{ВКС}}$. aerospace aircraft is latter/last step/stage of multistage orbital apparatus, moreover fuel/propellant of this step/stage in process of dispersal/acceleration is not expended/consumed (speed $V_{\text{к}}$ it is reached at the end of work of m accelerator). If VKS in the final trajectory leaves in orbit due to its own fuel/propellant, then the m accelerator in this case will accelerate/disperse VKS to the speed

$$V = V_{1k} - \Delta V_{B.C.}$$

Value $\Delta V_{B.C.}$ is defined as

$$\Delta V_{B.C.} = 9.81 J_{T.BKC} \ln \frac{1}{1 - \Delta \overline{G}_{T.BKC}} - \Delta V_{n.BKC} \qquad (11.50)$$

where $\Delta \overline{G}_{T.BKC} = \frac{\Delta G_{T.BKC}}{G_{BKC}}$ - over-all payload ratio of fuel/propellant of VKS, expended in process of injection into orbit; $\Delta V_{n.BKC} = f(V)$ - with V>6.5 km/s $\Delta V_{n.BKC} \approx 0$.

It must be noted that in this diagram (multistage apparatus) use of fuel/propellant of VKS for dispersal/acceleration on entry/leaving in orbit is inexpedient, since this increases weight of structure of VKS and substantially is decreased its maneuverability in open space.

Solution of problem for optimization of parameters of flight vehicle by minimum of value G_0 at assigned magnitude G_{BKC} is reduced to solution of system $dG_0/di_{1,\dots,k} = 0$ during known limitations (here i - parameter). A strict solution of this problem is very bulky, since the majorities of variables, which determine value G_0, are in turn the functions of the unknown parameters and characteristics of the separate steps/stages of flight vehicle, for example:

$$\Delta V_{n} = \Delta V_{n}(V_{c}, \overline{P}, K, \dots, \dots);$$

$$J_{T.C.H} = J_{T.C.H}(H, V, \overline{P}_0, c_p, \dots)$$ and so forth.

This problem in process of sketch design it is better to solve by approximation method, which considerably simplifies solution and at the same time it gives precision/accuracy necessary for sketch design.

Essence of method consists in the fact that in equation (11.49) variables are fixed/recorded. Matter is facilitated by the fact that some variables with a sufficient precision/accuracy can be determined on the basis of experiment of the design of identical flight vehicles, other variables with the deviation from the optimum prove to be insignificant error for the solution of problem as a whole.

For preliminary design it is possible to accept $\overline{G}^{*}_{т.у}=0.85-0.92$ (this value of load ratio on fuel/propellant they have upper stages of contemporary carrier rockets); $J_{т.у}=\text{const}$ (for example, for fuel/propellant H_2+O_2, $J_{т.у}\approx 450$ s);

$$\Delta V_{y1}=\ldots=\Delta V_{y_m}=\frac{V_{1к}-V_{ст\,у_1}}{m},$$

or $\Delta V_{y_1}=\ldots=\Delta V_{y_m}=\dfrac{V_{1к}-V_{ст.у.}-\Delta V_{ВКС}}{m}$ (if part of fuel/propellant of VKS is spent on dispersal/acceleration), where $V_{ст\,у_1}$ - speed of start of first accelerator (i.e. speed, which communicates carrier aircraft to accelerated/dispersed with it load).

Increase $\Delta V_{пi}=f(V_{ст\,i})$ is taken either from graph in Fig. 11.11 or from calculation of optimum trajectory of injection into orbit.

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Let us find number of all steps/stages of flight vehicle, after determining number of accelerators, since n=m+2.

Value $\bar{G}_{п.н}$ will be, obviously, greater, the greater percentage it will compose payload weight, concluded in orbit, from useful load, accelerated/dispersed with carrier aircraft (and respectively less percentage it will compose weight of accelerators). From this condition should be determined value m.

Criterion $\bar{G}_{п.н}$ can be represented as

$$\bar{G}_{п.н} = \bar{G}'_{п.н}(1 - \bar{G}_{c.н}),$$

where

$$\bar{G}'_{п.н} = G_{п.н}\, G_{н.с.н}, \quad \bar{G}_{c.н} = \frac{G_{c.н}}{G_{c.н} - G_{н.с.н}} = \frac{G_{c.н}}{G_0}.$$

Value $\bar{G}'_{п.н}$ - it is easy to determine from equation (11.49), which in this case will take form

$$\bar{G}'_{п.н} = \prod_{i=1}^{m} \frac{\bar{G}^{*}_{тi} - 1 - \dfrac{1}{e^{\frac{\Delta V - \Delta V}{9,81 \ldots}}}}{\bar{G}^{*}_{тi}}. \tag{11.51}$$

From equation (11.51) it is possible to determine, what part of load, accelerated/dispersed with carrier aircraft, is capable of leaving on near earth orbit.

Varying with speed of start of first accelerator and by Mach number, it is possible to determine appropriate values of value $\bar{G}_{п.н}$.

Fig. 11.20 shows graphical solution of equations (11.51).

Analyzing obtained dependence, it is possible to draw conclusion: for real values of parameters of accelerators increase in number m>2 virtually to increase in criterion $\bar{G}_{п.н}$ does not lead; therefore for multistage orbital apparatus should be accepted number of accelerators m=2 and number of all steps/stages, consequently, n=4.

Equation (11.49), which determines launching weight of multistage apparatus, in that case will take form

$$G_0 = \frac{\bar{G}^*_{т.с.н}}{\bar{G}^*_{т.с.н} - 1 + \frac{1}{e^{\frac{\Delta V_{с.н} - \Delta V_{п.с.н}}{9,81 J_{т.с.н}}}}} \, \frac{\bar{G}^*_{т.у_1}}{\bar{G}^*_{т.у_1} - 1 + \frac{1}{e^{\frac{\Delta V_{у_1} - \Delta V_{п.у_1}}{9,81 J_{т.у_1}}}}} \times$$

$$\times \frac{\bar{G}^*_{т.у_2}}{\bar{G}^*_{т.у_2} - 1 + \frac{1}{e^{\frac{\Delta V_{у_2} + \Delta V_{п.у_2}}{9,81 J_{т.у_2}}}}} \, G_{вкс}. \qquad (11.52)$$

Obviously, for each value $V_{ст.у_1}$ there is optimum value $V_{ст.у_2}$ determining optimum distribution of total weight of accelerators (between first and second accelerator). However, the error in the determination of the maximum of criterion $G_{п.н.}$ obtained during the replacement of optimum value $V_{ст.у_2}$ by recommended value

$$V_{ст.у_2} = V_{ст.у_1} + \Delta V_{у_1} = \frac{V_{1к} + V_{ст.у_1}}{2},$$

will be insignificant (Fig. 11.21).
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Thus, solving assigned mission by proposed approximation method, it is possible to find with sufficient precision/accuracy optimum value G_0 and series/row of important parameters of orbital apparatus, without resorting to determination of extremum of function of many variables.

Determining optimum weight of i step/stage from equation (11.48)

$$G_i=\left(\frac{\overline{G}^{*}_{\text{т}\,i}}{\overline{G}^{*}_{\text{т}\,i}-1+\dfrac{1}{e^{\frac{\Delta V_i+\Delta V_{n.\,i}}{9,81 J_{\text{т}.\,i}}}}}-1\right)G_{\text{н}.\,i} \qquad (11.53)$$

let us find weights of separate steps/stages of multistage flight vehicle.

If payload weight $G_{\text{вкс}}$ of that concluded in orbit is prescribed/assigned, and it is necessary to determine minimum launching weight G_0, then sequence of determining weight of separate steps/stages must be similar: weight of second accelerator; weight of first accelerator; weight of carrier aircraft.

From equation (11.53) we will obtain:

- weight of second accelerator

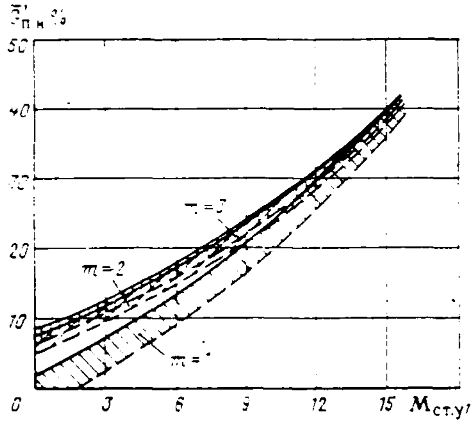
$$G_{y2}=\left(\frac{\overline{G}^{*}_{\text{т}.y_2}}{\overline{G}^{*}_{\text{т}.y_2}-1+\dfrac{1}{e^{\frac{\Delta V_{y_2}+\Delta V_{n.y_2}}{9,81 J_{\text{т}.y_2}}}}}-1\right)G_{\text{вкс}}; \qquad (11.54)$$

- weight of first accelerator

$$G_{y1}=\left(\frac{\overline{G}^{*}_{\text{т}.y1}}{\overline{G}^{*}_{\text{т}.y1}-1+\dfrac{1}{e^{\frac{\Delta V_{y1}+\Delta V_{n.y1}}{9,81 J_{\text{т}.y1}}}}}-1\right)(G_{y2}+G_{\text{вкс}}) \qquad (11.55)$$

- weight of carrier aircraft

$$G_{\text{с.н}}=\left(\frac{\overline{G}^{*}_{\text{т.с.н}}}{\overline{G}^{*}_{\text{т.с.н}}-1-\dfrac{1}{e^{\frac{\Delta V_{\text{с.н}}+\Delta V_{\text{п.с.н}}}{9{,}81 J_{\text{т.с.н}}}}}}-1\right)G_{\text{у1}}-G_{\text{у2}}+G_{\text{вкс}} \qquad (11.56)$$



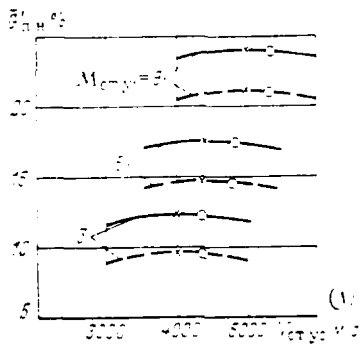


Fig. 11.20.   Fig. 11.21.

Fig. 11.20. Effect of number of accelerators and speed of start from carrier aircraft to value $\bar G'_{п.н}$ (fuel/propellant H_2+O_2, $J_{т.у}=450$ s):

———— $\bar G^*_{т.у}=0.9$, – – – $\bar G^*_{т.у}=0.85$

Fig. 11.21. Effect of speed of start of second accelerator on value $\bar G'_{п.н}$: ———— $\bar G^*_{т.у}=0.9$ – – – – $\bar G^*_{т.у}=0.85$, $\circ - V_{ст.у2}=\frac{V_{1к}-V_{ст.у1}}{2}$; $\times - V^{opt}_{ст.у2}$

Key: (1). m/s.

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From equation (11.56) it is possible to determine, what load and to what speed it is capable to drive away carrier aircraft

$$\bar G_{н.с.н}=\frac{G_{н.с.н}}{G_0}=\frac{\bar G^*_{т.с.н}-1-\frac{1}{e^{\frac{\Delta V_{с.н}+\Delta V_{п.с.н}}{9.81 J_{т.с.н}}}}}{\bar G^*_{т.с.н}}. \quad (11.57)$$

In order to solve system of equations (11.52)-(11.57), it is necessary to know value $\bar{G}^{\bullet}$ and value J_T for all working steps/stages. If for the accelerators value $\bar{G}_{T}$ is determined only by the perfection of construction/design and its possible value is actually known this load ratio on the fuel/propellant of upper stages of contemporary multistage rockets), then for the carrier aircraft value $\bar{G}^{\bullet}_{T.C.H}=G_{T.C.H}/G_{C.H}$ to determine in the stage of sketch design considerably more complicated. The necessary fuel load in this case will depend (besides the perfection of construction/design) on the characteristics of power plant, on the maximum speed of flight, on the parallax of orbit, on the conditions of takeoff, etc. Therefore the final value of value $\bar{G}^{\bullet}_{T.C.H}$ can be established only as a result of working design.

It is exactly the same also concerning specific impulse on fuel/propellant. For the power plant with ZhRD value $J_T=\text{const.}$ and its value for different fuels/propellants it is known. For the power plant of carrier aircraft with VRD value $J_{T.C.H}$ in the given equations is not true specific impulse on the fuel/propellant for VRD. In this case $J_{T.C.H}$ - required conditional pulse/momentum of VRD, i.e., this the specific impulse of conditional ZhRD, which, implementing the same work, would consume the same quantity of fuel/propellant (by the weight), as VRD. The values of the required true and conditional specific impulses of VRD do not coincide due to different value of the optimum thrust-weight ratio of aircraft with VRD and ZhRD. True specific impulse of VRD, defined as

$$J_{T.\text{ВРД}}=\frac{3600}{c_p},$$

must be considerably more than value $J_{T.C.H}$, in order to compensate large weight of VRD.

In period of preliminary design of value $\overline{G}^{*}_{т.с.н}$ and $J_{т.с.в}$ it is possible to find, after determining over-all payload ratio of fuel/propellant, required for dispersal/acceleration and climb, i.e., value $\overline{G}_{т.с.н}=\overline{G}_{т.с.н.}G_0$.

With precision/accuracy

$$\overline{G}_{т.с.в}=\frac{(H_{ст.у1}+V^2_{ст.у1}\,2g)\,c_{p\,ст}}{1300V_{ст.у1}}\,\frac{\overline{P}_0K_{ст}}{\overline{P}_0K_{ст}-1}, \qquad 11.58$$

sufficient for preliminary design where $H_{ст.у1}$ and $V_{ст.у1}$ - height/altitude and speed of start of first accelerator respectively in m and m/s; $c_{p.ст}$ - specific fuel consumption by engines of carrier aircraft at moment of start of first accelerator in kg/kg ·h; $\overline{P}_0=P_0/G_0$ - starting (takeoff) thrust-weight ratio of flight vehicle; $K_{ст}$ - lift-drag ratio of apparatus (aircraft with accelerated/dispersed load) at moment of start of first accelerator.

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For preliminary (or diploma) design it is possible to use data of Fig. 11.22.

Value $\overline{G}^{*}_{т.с.н}$ let us find, by solving equation of weight balance

$$\overline{G}^{*}_{т.с.н}=\frac{\overline{G}_{т.с.н}}{\overline{G}_{с.н.пуст}+\overline{G}_{т.с.н}}, \qquad 11.59$$

where $\overline{G}_{с.н.пуст}=\frac{G_{с.н.пуст}}{G_0}$; $G_{с.н.пуст}$ - weight of aircraft, carrier without accelerated/dispersed load and without fuel/propellant, spent on dispersal/acceleration to $V_{\text{с.т.}}$

For heavy supersonic and hypersonic aircraft

$$\bar{G}_{\text{т.н.п.ст}} = 0{,}3 - 0{,}4.$$

It is easy to determine required conditional pulse/momentum of VRD of carrier aircraft from equation (11.46), knowing value $\bar{G}_{\text{т.с.н}}$:

$$J_{\text{т.с.н}} = \frac{\Delta V_{\text{с.н}} + \Delta V_{\text{п.с.н}}}{9{,}81 \ln \frac{1}{1 - \bar{G}_{\text{т.с.н}}}}. \quad (11.60)$$

After determining thus for all steps/stages of flight vehicle of value $\bar{G}_{\text{т}}^{*}$ and $J_{\text{т}}$, it is possible to solve equations (11.52)-(11.57).

Fig. 11.23 and 11.24 show effect of speed of start of first accelerator to value $\bar{G}_{\text{н.с.н}} = \frac{G_{\text{н.с.н}}}{G_0}$ and $\bar{G}_{\text{п.н}} = \frac{G_{\text{ВКС}}}{G_0}$ (for above-recommended range of characteristics of carrier aircraft and accelerators).

Equations (11.52)-(11.57) for schematic of flight vehicle in question make it possible to determine optimum rate of dispersal/acceleration with the help of carrier aircraft (rate of start of first accelerator):

for cryogenic fuel/propellant $M_{\text{opt}} = 8-10$;

for hydrocarbon fuel $M_{\text{opt}} = 4-6$.

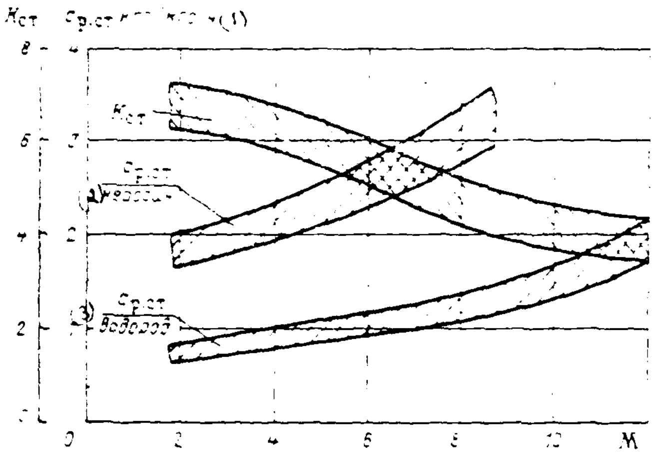

Fig. 11.22. Dependence of values $K_{ст}$ and $c_{p.ст}$ then Mach number.
Key: (1). kg/kg·h. (2). kerosene. (3). hydrogen.

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Payload weight. The weight of the true payload, concluded in orbit, is the most important factor during the design of orbital flight vehicle, obviously.

If we by payload of VKS understand weight of cosmonauts and transported cargo, then value $\overline{G}_{п.н ВКС}=G_{п.н ВКС}/G_{0 ВКС}$ can be determined, by using equation of weight balance

$$\overline{G}_{п.н ВКС}=1-\overline{G}_{ВКС пуст}-\overline{G}_{т ВКС},$$

where $\overline{G}_{ВКС пуст}=G_{ВКС пуст}/G_{0 ВКС}$ - over-all payload ratio of empty VKS (without load and fuel/propellant); $\overline{G}_{т ВКС}=G_{т ВКС}/G_{0 ВКС}$ - over-all payload ratio of complete reserve of fuel of VKS; $G_{0 ВКС}$ - weight of completely charged/filled VKS.

On value $\bar{G}_{\text{п.вВКС}}$ essential effect proves to be requirement for maneuverability in orbit (necessary fuel reserve for maneuver) and requirement for maneuverability in the atmosphere at hypersonic speeds (since on value of hypersonic lift-drag ratio K_r depends weight of structure of VKS).

During design of VKS (in the first approximation) it is possible to have following values of weight characteristics (see Table 11.2) in mind.

Fig. 11.25 shows value $\bar{G}_{\text{п.вВКС}}$ depending on hypersonic lift-drag ratio of apparatus.

Fuel load. Gross weight of the fuel/propellant of separate (i-th) step/stage in the general case will be defined as the sum

$$G_{\text{т}\Sigma}=G_{\text{т.р}}+G_{\text{т.м}}+G_{\text{т.в}}+G_{\text{т.н.з}},$$

where $G_{\text{т.р}}=G_{\text{т}i}$ - fuel load, required for dispersing/accelerating the load $G_{\text{н}i}$ (to value ΔV_i and ΔH_i); $G_{\text{т.м}}$ - the fuel load, required for maneuver accomplishment; $G_{\text{т.в}}$ - fuel load for the return to the base; $G_{\text{т.н.з}}$ - navigational fuel reserve.

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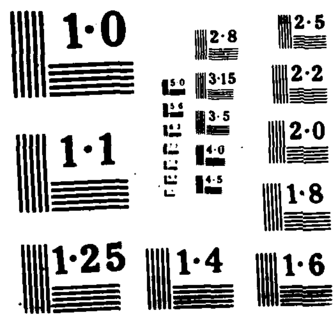
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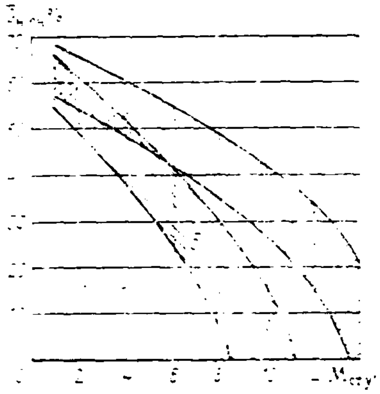

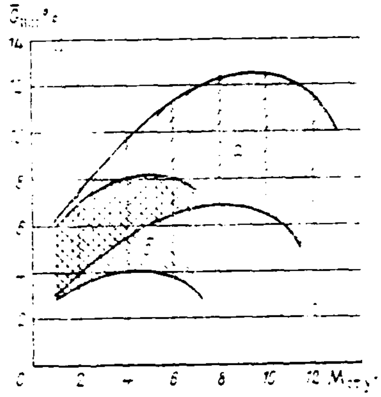

Fig. 11.23. Fig. 11.24.

Fig. 11.23. Dependence of over-all payload ratio of load, accelerated/dispersed with carrier aircraft with VRD, on rate of dispersal/acceleration: a) engines work on hydrogen; b) engines work on kerosene.

Fig. 11.24. Dependence of payload fraction, concluded in orbit, on rate of start of first accelerator: a) engines of all steps/stages work on hydrogen; b) engines of carrier aircraft work on kerosene, booster engines - on hydrogen.

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Let us consider components of complete fuel reserve for separate steps/stages of multistage orbital apparatus.

Knowing launching weights of separate steps/stages and value G^{*}_{T}, let us find fuel load, required for dispersing/accelerating load G_{H}:

$$G_{T.p}=\bar{G}^{*}_{T}G.$$

Virtually entire fuel/propellant of accelerators will be expended/consumed on increase in energy of accelerated/dispersed load; therefore for accelerators as one-time steps/stages $G_{т.м}+G_{т.в}+G_{т.н.з}=0$.

Fuel/propellant, required to carrier aircraft with VRD for maneuver accomplishment after starting/launching of load and for return on airport of departure in the first approximation, is equal (fuel/propellant - hydrogen)

$$G_{т.м}+G_{т.в}+G_{т.н.з}\approx(0.02-0.03)G_{с.н}.$$

If in process of injection into orbit fuel/propellant of VKS is not expended/consumed, then gross weight of fuel/propellant of VKS will be equal to

$$G_{т.ВКС}=G_{т.м}+G_{т.в}+G_{т.н.з}.$$

Fuel load, required for maneuver accomplishment in space, depends substantially on form of maneuver, which is in turn determined by value $\Delta V_м$ - by change in velocity vector. The relationship/ratio between value $\Delta V_м$ and required fuel load can be obtained from equation (11.45):

$$G_{т.м}=G_{ВКС}\left(1-\frac{1}{e^{\frac{\Delta V_м}{9,81 J_{т.ВКС}}}}\right). \qquad (11.61)$$

Here $G_{ВКС}$ - initial weight of apparatus in orbit (before the maneuver); $J_{т.ВКС}$ - specific jet firing of VKS, with the help of which is implemented the maneuver.

Table 11.2. Weight characteristics of VKS.

(1) Относительный вес	$K_г = 1,0$	$K_г = 2,3$
(2) Конструкция (с теплоизоляцией), оборудование, системы	0,52—0,37	0,55—0,41
(3) Двигательная установка	0,015—0,03	
(4) Шасси	0,02—0,035	
(5) Управление (аэродинамическое)	0,015—0,02	
(6) Управление (газоструйное)	0,01—0,015	
(7) Топливо	0,30—0,35	
(8) Полезная нагрузка (космонавты и грузы)	0,12—0,18	0,09—0,14

Key: (1). Overall payload ratio. (2). Construction/design (with thermal insulation), equipment, system. (3). Engine installation. (4). Chassis/landing gear. (5). Control (aerodynamic). (6). Control (gas-jet). (7). Fuel/propellant. (8). Payload (cosmonauts and loads).

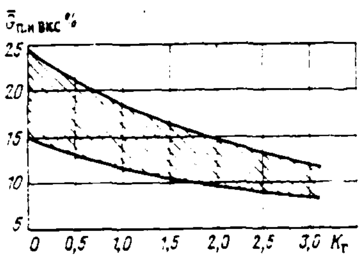

Fig. 11.25. Dependence of relative payload weight of VKS on hypersonic lift-drag ratio.

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Representation about numerical value of value $\Delta V_м$ and corresponding value $\bar{G}_{т.м}$ gives Table 11.3, in which are shown expenditures of fuel/propellant (when $J_{т.ВКС}$=450 s) on increase in altitude ΔH or on change in angle of slope Δφ of orbit.

Fuel/propellant for return in this case will be composed of three parts:

$$G_{т.в}=G_{т.сх}+G_{т.стаб}+G_{т.п.с},$$

where $G_{т.сх}$ - fuel load for orbit ejection (for creation of retro impulse); $G_{т.стаб}$ - fuel/propellant for stabilization (and administration) in initial stage of gliding/planning; $G_{т.п.с}$ - fuel/propellant for landing on prescribed/assigned airfield.

Value $G_{т.сх}$ is determined by equation (11.61), in this case it is necessary to have $\Delta V_{м}=\Delta V_{т}=30-70$ m/s (see § 2).

In period of preliminary design of VKS it is possible to accept:

$$G_{т.стаб}\approx 0.015-0.025\,G_{в.с};$$
$$G_{т.п.с}\approx 0.015-0.030\,G_{в.с};$$
$$G_{т.н.э}\approx 0.010-0.015\,G_{в.с}.$$

Required thrust-weight ratio of multistage apparatus. The starting thrust-weight ratio of multistage flight vehicle is defined as the relation

$$\overline{P}_0=\frac{P_0}{G_{с.н}-G_{н.с.н}}=\frac{P_0}{G_0},$$

where P_0 - total boost for launching of the engines of carrier aircraft.

In this case optimum value of value $\overline{P}_0$ must correspond to maximum of criterion $\bar{G}_{н.в}$. Therefore chosen the starting thrust-weight ratio of multistage flight vehicle should be in such a way as, other conditions being equal, to obtain the maximum value of value $G_{н.с.в.}$ and this can be done, after ensuring for the carrier aircraft the minimum of sum $(G_{т.с.в}+G_{дв.у})$.

So $G_{т.с.в}=f(\bar{P}_0)$ and $G_{дв.у}=\psi(P_0)$, then expressing values indicated through characteristics of aircraft and solving equation

$$\frac{d\bar{G}_{н.с.н}}{dP_0}=0,$$

we will obtain functional connection of optimum starting thrust-weight ratio with fundamental characteristics of carrier aircraft:

$$\bar{P}_0=\frac{1}{K_{ст}}+\sqrt{\frac{\left(H_{ст.у1}-\frac{V^2_{ст.у1}}{2g}\right)c_{p\,ст}}{1690V_{ст.у1}\gamma_{дв}K_{ст}}}, \qquad (11.62)$$

where $K_{ст}$ - and $c_{pст}$ (11.58) and Fig. 11.22; $\gamma_{дв}$ - weight per horsepower of power plant of carrier aircraft.

Table 11.3. Initial orbit: H=200 km; $V=V_{0}=7790$ m/s.

ΔH в км (1)	10	100	300	Δ [illegible] (1)	1	7	15
$\Delta V_{м} = \Delta V_{H}$ в м/с (1)	10	85	200	$\Delta V_{м} = \Delta V_{i}$ в м/с	140	950	2050
$G_{т.м}/G_{ВКС}$	0.002	0,02	0,05	$G_{т.м}/G_{ВКС}$	0,03	0,19	0.38

Key: (1). in the m/s.

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Starting thrust-weight ratio of accelerators and VKS will be defined as relation

$$\overline{P}_{y1} = \frac{P_{y1}}{G_{y1} - G_{н.\ y1}} = \frac{P_{y1}}{G_{н.\ с.н}};$$

$$\overline{P}_{y2} = \frac{P_{y2}}{G_{y2} - G_{н.\ y2}} = \frac{P_{y2}}{G_{н.\ y1}}; \quad \overline{P}_{ВКС} = \frac{P_{ВКС}}{G_{ВКС}}.$$

where P_{y1} - boost for launching of first accelerator; P_{y2} - boost for launching of second accelerator; $P_{ВКС}$ - boost for launching VKS.

If on accelerators and VKS ZhRD are established/installed, then thrust-weight ratio will not be limited with weight of engine installation, since specific weight of contemporary ZhRD is considerably lower than specific weight/gravity of VRD

$$\frac{\gamma_{дв\ ЖРД}}{\gamma_{дв\ ВРД}} \approx 0{,}05.$$

Therefore thrust-weight ratio of accelerators (without fearing overstress of engine installation) should be chosen from condition of guaranteeing acceptable g-force upon dispersal/acceleration and speed losses to gravitation and aerodynamic drag $\Delta V_{\text{п}}$, which, other conditions being equal, will determine value $\overline{G}_{\text{п.н}}$. Fig. 11.26 shows the effect of the starting thrust-weight ratio of the first accelerator to value $\overline{G}_{\text{п.н}}$ (for $M_{\text{с.т.у1}}=6$).

In period of preliminary design, taking into account possible g-limitations, it is possible to accept

$$\overline{P}_{y1} \approx \overline{P}_{y2} \approx \overline{P}_{\text{ВКС}} = 1.5-2.0.$$

Aerospace aircraft with YaRD [38].

Aerospace aircraft with nuclear rocket engine (YaRD), as it was shown above, can leave in orbit around Earth without aid of intermediate accelerator stage, and when $J_{\text{т.ЯРД}}>2000$ and without aid of carrier aircraft. The over-all payload ratio of fuel/propellant, required for injection into orbit of single-stage VKS, is determined by equation (11.46), which in this case will take the form

$$\overline{G}_{\text{т.пот}} = \frac{G_{\text{т.пот}}}{G_{0\,\text{ВКС}}} = 1 - \frac{1}{e^{\frac{V_{1\text{к}} - \Delta V_{\text{п}}}{9.81 J_{\text{т.ЯРД}}}}}. \quad (11.63)$$

Here $G_{0\text{ВКС}}$ - launching weight of VKS with YaRD (when $V_{\text{ст}}=0$); $\Delta V_{\text{п}}$ - cm. Fig. 11.11 (when $V_{\text{ст}}=0$).

The effect of $J_{\text{т.ЯРД}}$ on quantity $\overline{G}_{\text{т.пот}}$ is shown on Fig. 11.12 (when $M_{\text{ст}}=0$).

If we payload weight, concluded in orbit, consider weight of repeated flight vehicle when $V_{1\text{к}}$, then for single-stage VKS with YaRD payload will be equal to

$$G_{\text{п.н}} = G_{\text{ВКС}} = G_{0\,\text{ВКС}} - G_{\text{т.пот}}.$$

but value $\overline{G}_{п.н}$ is equal to

$$\overline{G}_{п.н}=\frac{G_{в-с}}{G_{с\,вх\,с}}=\frac{1}{e^{\frac{U_{к}+\Delta U_{р}}{9.81 U_{т.ЯР2}}}}. \qquad (11.64)$$

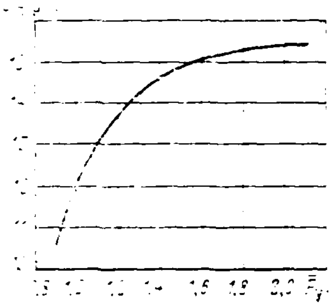

Fig. 11.26. Effect of starting thrust-weight ratio of first accelerator on value $\overline{G}'_{п.н.}$ $(M_{ст.у1}=6)$

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If payload for single-stage VKS is understood then just as for usual aircraft, then in that case payload will be determined by equation of weight balance, by solving which together with (11.63), it is possible to find over-all payload ratio of true payload, concluded in orbit, and consequently, and weight of single-stage VKS with YaRD

$$\overline{G}_{п.н} = \frac{G_{п.н}}{G_{0\,ВКС}} = \frac{1}{e^{\frac{V_{ik}+\Delta V_{п}}{9{,}81 J_{т.ЯРД}}}} - \overline{G}_{ВКС.пуст} + \overline{G}_{т.резерв}, \qquad (11.65)$$

where $\overline{G}_{т.резерв} = \overline{G}_{т} - \overline{G}_{т.пот}$ - over-all payload ratio of standby fuel/propellant.

In period of preliminary design of VKS with YaRD approximate value of relative empty weight (taking into account biological protection) can be taken in limits $G_{ВКС.пуст} = 0{,}40 - 0{,}65$.

Value $\bar{G}_{т.резерв}$ will be equal to

$$\bar{G}_{т.резерв}=\frac{G_{т.м}-G_{т.в}-G_{т.н.з}}{G_{0ВКС}}=\bar{G}_{т.м}-\bar{G}_{т.в}-\bar{G}_{т.н.з}$$

Fuel load for maneuver accomplishment in space in this case is also determined by equation (11.61), and value $\bar{G}_{т.в}$ and $\bar{G}_{т.н.з}$ in the first approximation, can be taken $\bar{G}_{т.в}+\bar{G}_{т.н.з}=0.03-0.06$. If single-stage VKS has the combined engine installation (VRD+YaRD) and is switched on YaRD when $V_{ст}\neq 0$, then the overall payload ratio of the true payload, concluded in orbit, analogous (with 11.65) will be determined then:

$$\bar{G}_{п.н}=\frac{G_{п.н}}{G_{0ВКС}}=\frac{1-\bar{G}_{т.ВРД}}{e^{\frac{V_{1к}+\Delta V_{п}-V_{ст}}{9.81 J_{т.ЯРД}}}}-\xi-\bar{G}_{т.резерв} \qquad (11.66)$$

Here $\bar{G}_{т.ВРД}=\frac{G_{т.ВРД}}{G_{0ВКС}}$ - overall payload ratio of fuel/propellant of VRD, required for arrival at altitude and speed, at which is switched on YaRD. Value $\bar{G}_{т.ВРД}$ is determined from equation (11.58), in which the moment/torque of the start of the first accelerator in this case is the moment of switching on YaRD; $\Delta V_{а}$ - corresponds to value $V_{ст}$ (see Fig. 11.11)

$$\xi=\bar{G}_{ВКС.пуст}+\bar{G}_{дв.у.ВРД},$$

where $\bar{G}_{дв.у.ВРД}=G_{дв.у.ВРД}/G_{0ВКС}\approx 1.3\gamma_{дв}\bar{P}_{0ВРД}$ [coefficient of 1.3 considers the weight of air intakes, air ducts, etc.]. Starting thrust-weight ratio $\bar{P}_{0ВРД}=P_{0ВРД}/G_{0ВКС}$ is determined by equation (11.62).

Finally, if VKS has YaRD with $J_{т.ЯРД}<2000$ and for injection into orbit is required carrier aircraft, which accelerates/disperses VKS to necessary value $V_{\text{ст}}$, then this flight vehicle can be considered as multistage with m=0. Due to its own fuel/propellant VKS must increase rate by value $\Delta V_{\text{BKC}}=V_{1\text{к}}-V_{\text{ст}}$.

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Gross weight of fuel/propellant of VKS in this case will be equal to $G_{\text{т.ВКС}}=G_{\text{т.п}}+G_{\text{т.м}}-G_{\text{т.в}}-G_{\text{т.н.з}}$, where $G_{\text{т.п.т}}$ - fuel load, required for injection of VKS in orbit after start from the carrier aircraft [see (11.46)]. All parameters of two-stage flight vehicle are determined by the dependences given above.

Special features of the selection of geometric parameters of VKS.

On geometry of VKS have simultaneous effect of requirements, presented to flight characteristics during hypersonic gliding/planning, and requirements, presented to landing data of apparatus. Each requirements in the final analysis are expressed by the value of the hypersonic and subsonic lift-drag ratio (takeoff data in this case have smaller value, since for the takeoff of aerospace flight vehicle to it is more expedient use special rocket dolly).

Important effect on geometry of VKS will prove to be aerodynamic reentry heating. The solution of problem requires compromises between the geometry at the hypersonic and subsonic flight speeds. A change in the hypersonic configuration affects subsonic characteristics and vice versa. This interdependence and incompatibility of characteristics for the hypersonic and subsonic flight substantially complicates aerodynamic development and requires often the application of the variable geometry. Therefore high value in the design of VKS obtained the idea of the "distribution of hypersonic and subsonic regimes". The configuration of the apparatus, planned according to this principle facilitates the problem of aerodynamic investigations and provides reaching/achievement of the subsonic flight characteristics, close to the usual aircraft.

As example of VKS with divided flight conditions it is possible to serve design of apparatus VL-3A (Fig. 11.27).

This VKS has put forth wings for distribution of hypersonic and subsonic regimes. The hypersonic lift-drag ratio of apparatus is equal to K_r=2.3, at subsonic speed with the advanced wings K=8.

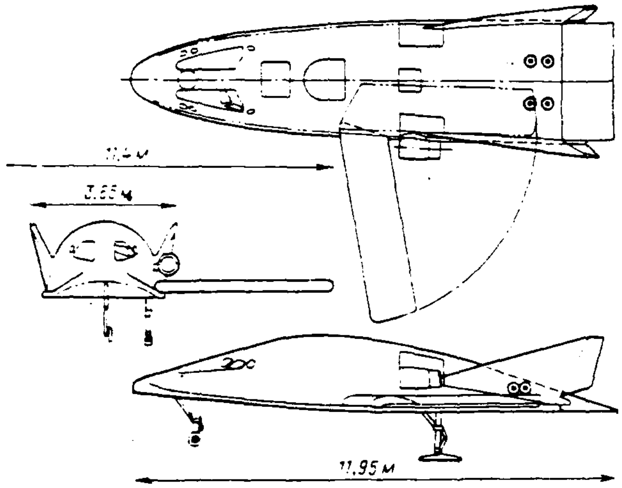

Fig. 11.27. Diagram of VKS VL-3A.

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After completion of hypersonic phase of gliding/planning wings partially are put forth for positioning center of pressure in transonic region. At subsonic speed the wings are put forth completely. Two TRD, which ensure aircraft landing, are put forth and are started. There is a reserve of fuel (to 10 min flight with the full thrust) for the approach guidance, "tightenings" and the departure to the second circle. For crash landing and splashdown is used parachute. Landing shock with the parachute is absorbed by the tail section of the apparatus, which concerns the earth/ground of the first.

Specific wing load or on lifting surface (for apparatuses with lifting body) is one of most important geometric parameters of flight vehicles. In this case by lifting surface is understood the projected area of apparatus in the plan/layout. For VKS during the hypersonic flight the specific load on the lifting surface will depend on equation (11.1), consequently, the value of this parameter, necessary for the equilibrium flight (gliding/planning) at the given rate at the given height/altitude, can be determined from equation (11.4)

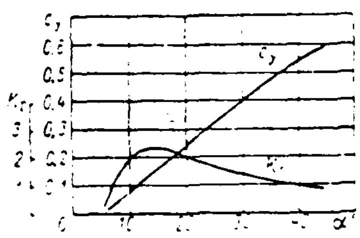
$$p=\frac{31\cdot10^{6}\rho V^{2}_{\text{г.п}}c_{y}}{62\cdot10^{6}-(V_{\text{г.п}}\pm460\cos\varphi)^{2}}. \qquad (11.67)$$

Values $c_y(\alpha)$ and $K_r(\alpha)$ are given in Fig. 11.28. For calculating the load on the m² of surface of VKS should be accepted c_y, appropriate $K_{r\,max}$.

Task of selection of specific wing load, necessary for obtaining of acceptable subsonic characteristics of VKS (mainly landing), does not have vital differences from analogous task for usual aircraft. Fig. 11.29 gives an example of subsonic aerodynamic characteristics of VKS with lifting body.

On exterior form of VKS essential effect proves to be aerodynamic heating. Heat transfer rate, which enters the skin/sheathing from the boundary layer, as is known, it is proportional to the local coefficient of convection heat transfer on the boundary air-skin/sheathing [see equation (11.35)], which depends substantially on the sweepback of the body (with an increase in the sweepback the coefficient of heat transfer sharply it is reduced, Fig. 11.30).

Therefore all protruding into the flow parts of VKS must have the large sweepback (from these considerations the managers of surface, consequently, there must not deviate to the large angles). Sweep angle should be accepted not less than 70-75°.



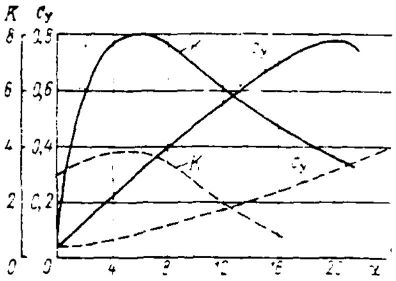


Fig. 11.28. Fig. 11.29.

Fig. 11.28. Hypersonic aerodynamic characteristics of VKS with lifting body in trim position center of gravity (apparatus VL-3A, V=6080 m/s; H=60.8 km).

Fig. 11.29. Subsonic aerodynamic characteristics of VKS with lifting body (apparatus VL-3A; $S_{кр}$ =28.1 m²; $S_{кр}$ =9.7 m²): ——— - by wing; - - - without wing.

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Coefficient of convection heat transfer near critical point depends on radius of nose section of body. Solving together the equation, which describes heat transfer by thermal cycling, and the equation, which describes aerodynamic heating at critical point on the sphere, it is possible to obtain the approximate dependence, which links a radius of forebody of VKS with the parameters of the hypersonic flight

$$r=\frac{0,00019\cdot q\,[(V/1000)^2+0,67T_{пов}/1000]^2}{\varepsilon^2(T_{пов}/1000)^8}, \qquad (11.68)$$

where r - radius of nose section m; q - velocity head in the undisturbed flow in kgf/m²; V - flight speed in the m/s; $T_{пов}$ -

temperature of external surface in °K; ϵ - coefficient of thermal cycling material of nose section.

During design of exterior form of VKS usually are examined two systems of heat shielding from aerodynamic heating - ablation and radiation. It is considered that for the apparatuses with $K_{r}<1.5$ the advantage in proportion by weight has the ablation system of heat shielding, whereas for the apparatuses with the higher value of hypersonic lift-drag ratio more complex radiation system is required. The combined heat shielding, when ablation system is used only for the protection of lower surface of VKS, deserves considerable attention. The removable lower heat shield, covered with ablating material, defends well construction/design of VKS from high temperatures on the lower surface.

System of heat shielding of VKS VL-3A (Fig. 11.31 and 11.32) can serve as example of combined system of heat shielding.

Recommendations given in present chapter are not, certainly, by comprehensive material according to design of aerospace aircraft.

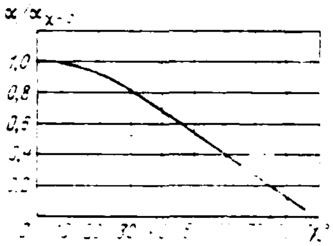

Fig. 11.30. Dependence of coefficient of heat transfer in zone of leading edge on sweep angle.

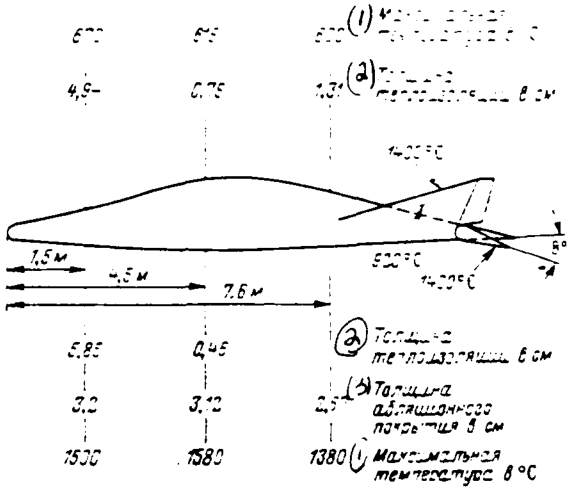

Fig. 11.31. System of heat shielding apparatus LV-3A (ablation coating - purple mixture NASA, density of 650 kG/m³; thermal insulation - micro-quartz, density of 70 kG/m³.

Key: (1). Maximum temperature in °C. (2). Thickness of thermal insulation in cm. (3). Thickness of ablation coating in cm.

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Nevertheless they allow in the period of the preliminary design of apparatus to solve many important problems.

For example, it is necessary to plan multistage flight vehicle, capable of delivering to near earth orbit $H \approx 150$ km 10 cosmonauts and 500 kg of load ($G_{г.вВКС}=1500$ kg). The latter/last step/stage of apparatus - VKS must increase orbit altitude to 500 km and change orbit inclination by the angle to 7°. With the return, in the process of hypersonic gliding/planning, VKS must fly not less than 16500 km and accomplish lateral maneuver, reaching in this case lateral distance $L_{п.бок}=4000$ km. Maximum equilibrium temperature on the surface of VKS, not defended by ablation coating, must not exceed 1400°C, and the time of action of temperature t≥1000°C must not exceed 70 min. It is necessary to determine the optimum values of the basic parameters and the flight characteristics of VKS and the multistage flight vehicle as a whole. As the fuel in the engines of all steps/stages to take liquid hydrogen.

Using given in this chapter formulas and graphic dependences, we find:

a) aerospace aircraft

$$G_{0\,ВКС}=12 \text{ тс}^{(1)};\ \overline{G}_{п.н.\,ВКС}=0,125;\ P_{ВКС}=20 \text{ тс}^{(1)};$$
$$G^{т}_{т.ВКС}=3520 \text{ кгс}^{(2)};\ \overline{G}_{т.ВКС}\approx 0,3;\ S_{ВКС}=52 \text{ м}^2$$
$$p_{ВКС}=230 \text{ кгс/м}^2\,^{(3)};\ K_г=2;\ V_{1к}=7810 \text{ м/с}^{(4)};$$
$$\Delta V_H=109,4 \text{ м/с}^{(4)};\ \Delta V_{\varphi}=954 \text{ м/с}^{(4)};$$

Key: (1). t. (2). kg. (3). kgf/m². (4). m/s.

b) second accelerator

$$G_{y2}=11{,}9 \text{ тс}; \; P_{y2}=41 \text{ тс}; \; G_{т.y2}=10{,}7 \text{ тс};$$
$$V_{ст.y2}=5255 \text{ м/с}; \; \Delta V_{y2}=2555 \text{ м/с};$$

Key: (1). t. (2). m/s.

c) first accelerator.

$$G_{y1}=29{,}2 \text{ тс}; \; P_{y1}=90 \text{ тс}; \; G_{т.y1}=24{,}9 \text{ тс};$$
$$V_{ст.y1}=2700 \text{ м/с}; \; \Delta V_{y1}=2555 \text{ м/с};$$

Key: (1). t. (2). m/s.

d) hypersonic carrier aircraft

$$G_{с.н}=77{,}7 \text{ т}; \; P_{с.н}=P_0=76 \text{ тс}; \; G^{\Sigma}_{т.с.н}=33{,}8 \text{ тс};$$
$$\overline{G}_{т.с.н}=0{,}244; \; \overline{G}^{\bullet}_{т.с.н}=0{,}411; \; J_{т.с.н}=1422 \text{ с}; \; M_{\max}=9;$$

Key: (1). t. (2). s.

e) multistage flight vehicle

$$G_{[illegible]}=13[illegible].8 \text{ тс при } m=1; \; G_0=175 \text{ тс};$$
$$\overline{G}_{[illegible]}=G_{[illegible]}/G_0=0{,}0917; \; \overline{P}_0=0{,}58 \text{ и т. д.}$$

Key: (1). t. (2). with. (3). and so forth.

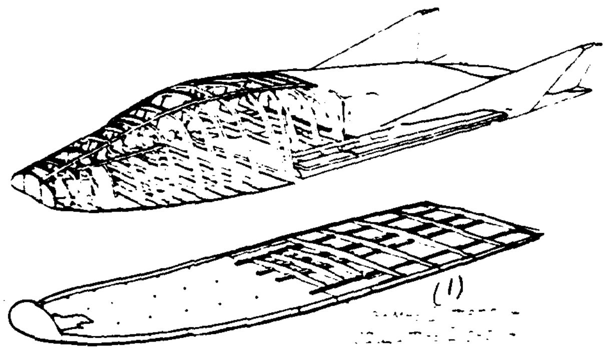

Fig. 11.32. Construction/design of VKS VL-3A.

Key:(1).Removable heat shield.

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Chapter XIX.

DESIGN OF CONTROL SYSTEMS OF AIRCRAFT.

Process of change in time of forces acting on aircraft and moments/torques for obtaining necessary flight trajectory is called control, and set of devices/equipment, which ensure this process, system of control of aircraft. This system is frequently called also the *main* or *primary system* for control, since besides it on the aircraft there are systems of synchro control, which ensure control of other different devices/equipment: aerodynamic trim tabs, adjustable with stabilizer, landing gear lowering and retracting, the brakes of landing gear wheels, the turn of front landing gear strut with the taxiing, the high-lift device of wing (flaps and slats), speed brakes, shutters/doors of hatches and so forth, etc.

Present chapter is dedicated to design of primary system for control.

Design of systems of synchro control, as a rule, does not represent independent task and is accomplished/realized in process of designing electro- and hydraulic systems of aircraft.

§1. Control loop, its fundamental elements.

Control of aircraft can be accomplished/realized either by pilot or automatic systems. The latter received wide acceptance at present and solve diverse problems from the simple maintaining of the flight conditions (simplest autopilots) prescribed/assigned by pilot to the interception of air targets, landing approach and so forth, etc.

During control of pilot change in position of aircraft in space (on height/altitude, attitudes of roll, pitch and yaw), and also change in flight conditions (on speed, g-force, angles of attack and slip) is usually determined by pilot visually along terrestrial reference points and line of horizon, while in the absence of visibility - according to flight and special instruments (on speed indicators, g-force, angles of attack, bank and slip, altimeter, rate-of-climb indicator, etc.).

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Physical sensations of g-forces and change in force feel play a large role in information of pilot about change in flight conditions and about action of controls. The comparison of the instantaneous values of the parameters of flight with the required is allowed for pilot on the basis of the specific skills on the piloting to form/shape control signals in the form of the displacements/movements of control levers. These displacements/movements of levers are converted by control system into the appropriate control displacements (controls) of aircraft. The control displacement leads to the fact that the aircraft begins to vary the parameters of flight. With the help of the sense organs the pilot checks the results of his effects on the controls (i.e. a change in the parameters of the flight of aircraft), attempting to reduce the disagreement/mismatch between their current and required values, and it ceases effect on the control levers, if these values correspond to each other.

Thus, in flight is formed closed control loop, which consists of three fundamental connected with each other elements (components/links): pilot, system of control and aircraft (Fig. 19.1a). The characteristics of this duct/contour and its stability are defined by the characteristics of the fundamental entering it elements (i.e. pilot, the system of control and aircraft as the object of control), by the interconnection of these elements and by their congruence. Therefore the success of accomplishing flight mission in many respects depends on how are successfully selected the characteristics of the circuit elements of control.

As circuit element of control, pilot, in turn, simply can be considered as automatic control system, which is formed by three fundamental connected elements (Fig. 19.1b): by sense organs (receptors - "sensors"); by central nervous system, which fulfills functions of processing information and making of decision, and by actuating elements (muscles of hands, legs, back). The motions of the actuating elements of pilot and the efforts/forces developed by them are the "output signals" ("output") of man as the component/link of control loop. In this case the information about the action of actuating elements (the muscles) is transmitted to the central nervous system through kinestatic receptors, which accomplish/realize feedback in the organism, similarly to feedback in the automatic control systems.

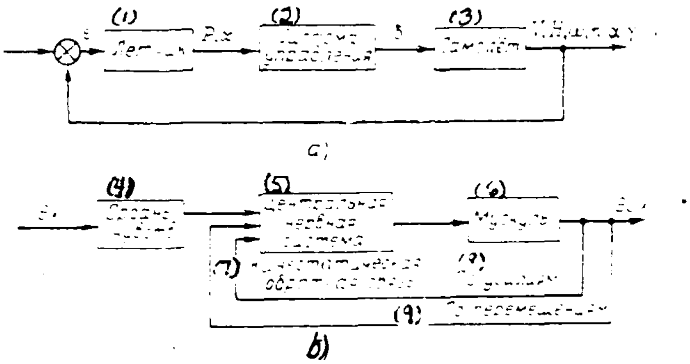

Fig. 19.1. Diagram of closed control loop "pilot - system of control - aircraft" (a) and structural diagram of operator-pilot as component/link of control loop (b): ϵ - disagreement/mismatch between current and required values of parameters of flight; P, x - forces, accompanying control levers and their displacement/movement; δ - control displacement; V, H, ω, n, α, γ - parameters of flight; вх - input perceptions (visual, auditory, sensation of accelerations); вых - output effects on control levers.

Key: (1). Pilot. (2). Control system. (3). Aircraft. (4). Sense organs. (5). Central nervous system. (6). Muscles. (7). Kinestatic feedback. (8). On efforts/forces. (9). On displacements/movements.

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These internal connections in the organism of man-pilot are called kinestatic feedback, because of which the system "regulator - pilot" is closed, and proves to be possible dosing the motions of actuating elements both in the extent of movements and over efforts/forces developed in this case.

Possibility of dosing efforts/forces and displacements of actuating elements is allowed for pilot, familiar with handlings of this aircraft, to accomplish/realize required changes in flight conditions by corresponding displacement of control levers during application to them of specific efforts/forces.

In this case one should note that change in efforts/forces is better perceived by pilots than displacement, and precision/accuracy of dosing efforts/forces substantially higher than precision/accuracy of dosing displacements. Therefore among the handling characteristics handlings on the efforts/forces have larger value. If pilot does not perceive changes in forces on control levers with a change in the flight conditions (which can occur, for example, during the fully powered controls without appropriate loaders), then it is deprived of the very important part of the information, but, system, "regulator - pilot" proves to be extended on one of the parameters of feedback. As a rule, this leads to the very large errors in control, corollary of which can be the loss of stability of entire control loop, which is expressed in the form of the progressive "rocking" of aircraft.

As component/link of control loop pilot possesses a series of properties, which affect control process. From these properties it is possible to note:

1. Delay of response reaction to the external signals. This value strongly depends on the trained state of pilot and his psychological and physical state. For the normal conditions it is possible to accept the time lag of the reaction of the pilot of average/mean qualification 0.2-0.3 s.

2. Certain dead zone.

3. Capability for filtration of external signals.

4. Capability for change of its own transfer function over wide limits, including differentiation and integration, i.e., capability for reaction not only to deviations of any parameter of flight from required value, but also for first and second derivatives (speed and acceleration) this deviation, and also for integral of this deviation.

5. Capability for shaping of output control signals with specific limited precision/accuracy, which depends on value of these signals and their frequency.

6. Ability of tracking signals, which enter with frequency is not more than 2.5-3 Hz (presence of passband).

In spite of capability for change of its own transfer function over wide limits, most accurately pilot works to control loop as single-channel amplifier, which consecutively/serially removes disagreement/mismatch with respect to any parameter. Taking into account this, it is necessary so to project/design the remaining circuit elements of control so that in the portion of pilot would fall precisely this simplest function - amplification.

S2. Aircraft as the object of control.

Examining control loop, one cannot fail to stop at special features of characteristics of object of control - aircraft [1].

FOOTNOTE [1]. These characteristics in sufficient detail are examined in other courses ("flight dynamics", "stability and aircraft handling"). ENDFOOTNOTE.

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Aircraft in space has six degrees of freedom, and its motion is described by six by differential equations of motion (Euler). The solution of these equations in the general case would make it possible to determine the character of the spatial motion of aircraft at any moment of time and, in particular, after the effect of pilot on the controls, and also it would make it possible to judge the stability of this motion. However, the direct solution of these equations presents known difficulties even during the application of contemporary analog (continuous) computers (integrators). But if we for the reference-flight conditions take the rectilinear steady coordinated flight and to consider the deviations of the parameters of motion from the original values sufficiently small (to use the method of "slight disturbances"), then because of the symmetry of aircraft the system of six equations of motion can be divided into two independent systems of equations, with the known degree of accuracy of those describing the motion of aircraft in the plane of symmetry (the so-called "longitudinal" motion) and in other two planes ("lateral" motion).

During use of equations of kinematic constraints each of motions (longitudinal and lateral) is described by system of four differential equations. The system of four equations of axial motion describes two oscillatory motions, that develop after the cessation/discontinuation of effect on the aircraft of external disturbance/perturbation (atmospheric effect, the control displacement, power change, etc.). One of these motions (called short-period) is developed it rapidly and has the small period (order 1-5 s), another - is developed it comparatively slowly and has substantially larger period (order of several tens of seconds). This motion is called long-period (Fig. 19.2).

Solution of system of equations of yawing motion shows that yawing motion, as a rule, represents sum of two aperiodic and one oscillatory motion.

Long-period axial motion is easily parried by pilot and is not of special interest for research. Aperiodic yawing motions, of which one presents the rapidly damped roll, and another - sufficiently slowly developing "spiral" motion; also it does not represent special interest, since is not proven to be serious effect on the evaluation by the pilots of aircraft handling.

Another matter - short-period longitudinal and vibratory/oscillatory yawing motion. The character of the development of these motions after the action of external disturbances/perturbations or deflection of controls to the decisive degree is evaluation criterion by the pilots of stability and aircraft handling.

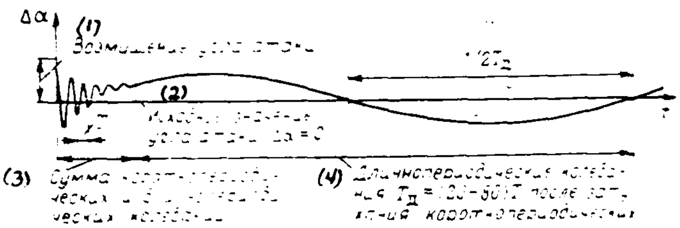

Fig. 19.2. Character of change of angle of attack of aircraft in process of attenuation of disturbed motion after action of external disturbance/perturbation of wind gust: T - period of short-period oscillations; T_2 - period of long-period oscillations.

Key: (1). Disturbance/perturbation of angle of attack. (2). Original value of angle of attack (.Δα=0). (3). Sum of short-period and long-period oscillations. (4). Long-period ... after attenuation of short-period.

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Therefore these motions by the most detailed form are investigated in the process of designing the aircraft in order to previously work out the appropriate measures in the case, if the parameters of these motions will prove to be unsatisfactory. At the stage of the design of aircraft these motions are investigated with the help of the electronic analog computers.

Let us examine in more detail equation of those agitated of longitudinal short-period and lateral vibratory/oscillatory of motions, since character of latter is completely determined by coefficients with corresponding derived parameters of motion, and these coefficients to a considerable degree can be changed by design engineers of aircraft via selection of its forms, centering and automation of control system.

Simplified equation, which describes change in angle of attack in longitudinal disturbed short-period motion, in dimensionless form takes form

$$\frac{d^2\Delta\alpha}{d\bar{t}^2}-\left(c_y^{\alpha}-\frac{m_z^{\bar{\omega}_z}-m_z^{\dot{\bar{\alpha}}}}{\bar{r}_z^2}\right)\frac{d\Delta\alpha}{d\bar{t}}-c_y^{\alpha}\frac{\mu\left(m_z^{c_y}-\frac{m_z^{\bar{\omega}_z}}{\mu}\right)}{\bar{r}_z^2}\Delta\alpha=0. \quad (19.1)$$

If this equation is written for the case, when the disturbance/perturbation of reference-flight conditions occurs as a result of the deflection of elevator $\Delta\delta_{B}$, then in the right side appears expression $\frac{\mu}{\bar{r}_z^2}m_z^{\delta_B}\Delta\delta_B$. Let us note that the relationship/ratio between changes in vertical g-force Δn_y and of angles of attack $\Delta\alpha$ has the form

$$\Delta n_y=\frac{q}{G/S}c_y^{\alpha}\Delta\alpha=\frac{c_y^{\alpha}\Delta\alpha}{c_{y\,r.f.}}. \quad (19.2)$$

In equations (19.1) and (19.2)

$\Delta\alpha$ - change of angle of attack in disturbed motion from its value in reference-flight conditions;

$$c_y^{\alpha}=\frac{\partial c_y}{\partial\alpha};\quad m_z^{\bar{\omega}_z}=\frac{\partial m_z}{\partial\bar{\omega}_z};\quad m_z^{\dot{\bar{\alpha}}}=\frac{\partial m_z}{\partial\dot{\bar{\alpha}}};$$

$\bar{r}_z^2 = \frac{I_z}{m b_a^2}$ - dimensionless radius of inertia of aircraft relative to axis/axle Oz_1;

$\bar{\omega}_z = \omega_z b_a / V$ - dimensionless rate of pitch;

$\bar{\dot{\alpha}} = \frac{\partial \alpha}{\partial t} \frac{b_a}{V}$ - dimensionless rate of change of angle of attack during rotation of aircraft;

$\bar{t} = \frac{t}{\tau}$ - dimensionless time, where $\tau = \frac{2m}{\rho S V}$.

m - mass of aircraft;

$\mu = \frac{2m}{\rho S b_a}$ - relative density of aircraft;

$m_z^{c_y} = \frac{\partial m_z}{\partial c_y} = \bar{x}_T - \bar{x}_F$ - longitudinal stability factor with respect to g-force;

$m_z^{\delta_B} = \frac{\partial m_z}{\partial \Delta \delta_B}$ - elevator-effectiveness derivative;

$c_{y\,r.n} = \frac{2G}{\rho S V^2}$ - value c_y in level flight.

It is known that in differential second order equation, [line missing in original text] (i.e. degree of damping) it is determined by coefficient value with first-order derivative (i.e. in this case when $\frac{d\Delta\alpha}{d\bar{t}}$, and frequency of these -- by coefficient value with zero derivative (i.e. with Δα).

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The stability condition of this motion, i.e., the condition for its attenuation, will be the positive value of these coefficients:

$$\left(c_y^\alpha - \frac{m_z^{\dot{\alpha}} - m_z^{\bar{\omega}_z}}{\bar{r}_z^2} \right) > 0; \quad -c_y^\alpha \frac{\mu \left(m_z^{c_y} - \frac{1}{\mu} m_z^{\bar{\omega}_z} \right)}{\bar{r}_z^2} > 0. \qquad 19.3$$

The sum

$$m_z^{c_y} + m_z^{\omega_z} \mu = \sigma_n \quad (19.4)$$

is called the coefficient of longitudinal static stability with respect to the g-force and plays very significant role in the formation of the most important longitudinal-behavior characteristics and controllability. Since always $c_y^{\alpha} > 0$, then from (19.3) it follows that for the stabilization of aircraft it is necessary, in order to $\sigma_n < 0$.

Expression (19.4) shows that even when $m_z^{c_y} = \bar{x}_T - \bar{x}_F > 0$ design engineers of aircraft have another possibility to ensure stability of aircraft due to increase in absolute value of derivative $m_z^{\omega_z}$ (since always $m_z^{\omega_z} < 0$). The latter can be achieved/reached, in particular, by introduction to the system of control of the damper of longitudinal oscillations (pitch damper).

In contemporary supersonic aircraft in comparison with subsonic all aerodynamic and inertial coefficients, entering equation (19.1) of longitudinal short-period motion, substantially were changed.

Thus, moment of inertia of aircraft relative to transverse axis J_z strongly increased due to considerable fineness ratio of fuselage and separation along their length of sufficiently large masses (equipment, engines, fuel/propellant). Derivatives c_y^{α} and $m_z^{\omega_z}$ (and, consequently damping) substantially decreased due to the application of thin sweptback wings and tail assembly, and also due to the special features of aerodynamics at supersonic speeds. Longitudinal stability factor with respect to the g-force with M>1 considerably grows due to the shift/shear of focus back/ago.

All this leads to adverse change in characteristics of dynamic stability and controllability, by which are understood transient-response characteristics of return to reference-flight conditions after action on aircraft of external disturbance/perturbation or setting of new values of parameters of flight after control displacement (Fig. 19.3b). Fig. 19.3a shows the exemplary/approximate character of the transient processes of changing the normal load factor with the deflection of elevator at the subsonic and supersonic flight speeds (by dotted line shown a change in the transient process at supersonic speed during the introduction of the damper of the longitudinal oscillations of aircraft).

From examination of Fig. 19.3a it is evident that at supersonic speeds (at high altitudes) deflection of elevator leads to intense initial excess/overshoot and for long sustained oscillations with high frequency of controlled parameter [in case - angle of attack in question and, consequently, vertical g-force, connected with relationship/ratio (19.2)]. To control/guide aircraft with such characteristics is extremely difficult, and sometimes also it is impossible, since pilot, having the characteristics (delay and inertness) characteristic to him, attempting to extinguish the oscillations, begins "to swing" aircraft.

Phenomenon of "rocking" of aircraft by pilot, that consists in rapid increase in amplitude of oscillations of aircraft and testifying about loss of stability of control loop, was encountered in flying practice in the beginning of mastery/adoption of high flight speeds, until designers of control systems accept special measures.

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Correctly planned aircraft always possesses longitudinal static stability and longitudinal damping [see conditions (19.3)]. Therefore in any flight conditions the emergent as a result of disturbance/perturbation short-period oscillations of aircraft, if pilot does not attempt to extinguish them, compulsorily attenuate (true, in some flight conditions for this can be required sufficiently long time !) aircraft is returned to the initial regime (if disturbance/perturbation was caused by external reason) or is balanced in the new regime (if disturbance/perturbation it was caused by the deviation of altitude control).

Stability of aircraft as dynamic system is clearly illustrated by Fig. 19.4, where are given amplitude-phase frequency characteristics of some aircraft, on which in process of flight tests were fixed cases of "oscillation". As it follows from the figure, these aircraft in all flight conditions are stable. Depending on flight conditions vary only the factors of amplification, damping (damping decrements), natural frequencies, cutoff frequencies, etc.

But if pilot attempts to stabilize aircraft and to extinguish his oscillations by motions of control levers, then, interfering in control, he connects to stable oscillating circuit/member (aircraft as object of control) two additional components/links: system of control (transmitting component/link) and himself (controlling or command component/link), closing, thereby, control loop (see Fig. 19.1a). In this case on the frequency characteristics of aircraft (stable very on themselves) the frequency characteristics of the system of control and pilot himself are placed. From this the stability of control loop deteriorates (let us recall that, for example, at the frequency 1 Hz only its own temporary/time pilot's time delay 0.2 s creates phase lag - 72°, and with the delay 0.3 s - 108°), as a result in some flight conditions (for example, under conditions with the increased efficiency of controls or with underdamping) control loop can become unstable, begins "oscillation". First and radical combat means with it - "to disconnect" from the stable aircraft of pilot and control system, i.e., to jam control lever. In this case the aircraft will be quieted itself.

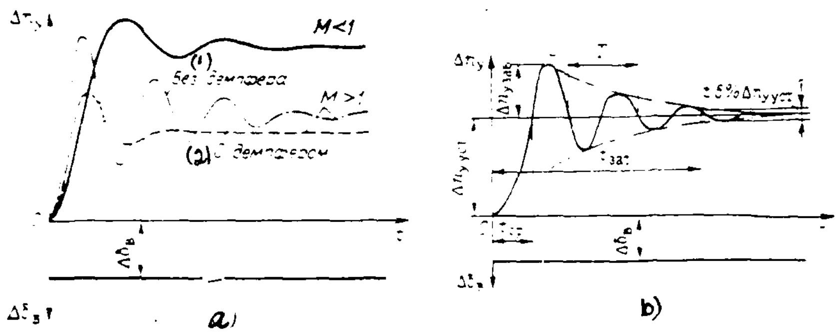

Fig. 19.3. Transient processes of changing g-force (Δn_y) in longitudinal short-period motion of aircraft after deflection of elevator ($\Delta\delta_в$): a) change in character of longitudinal short-period motion at subsonic and supersonic flight speeds, and also upon start of pitch damper (when Δ_ρ- b) indices, which characterize dynamic stability and aircraft handling; T - period of natural short-period oscillations; t_{cp} - triggering time; $\Delta n_{y\,заб}$ - initial excess/overshoot of controlled parameter (in this case - g-force); $\Delta n_{y\,уст}$ - conservative value of change in controlled parameter; $\Delta n_{y\,заб}/\Delta n_{y\,уст}$ - relative initial excess/overshoot - overcontrol; $t_{зат}$ - time of delay of transient process; $n_{зат}$ - number of oscillatory periods to damping moment.

Key: (1). Without the damper. (2). With damper.

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Let us consider now lateral oscillatory motion based on example of simplified differential equation of change in slip angle in disturbed motion (in dimensionless form). This equation takes the form

$$\frac{d^2\beta}{d\bar{t}^2}-\left(\bar{m}_y^{\bar{\omega}_y}-\frac{1}{2}c_z^\beta\right)\frac{d\beta}{d\bar{t}}-\left(\mu\bar{m}_y^\beta-\frac{1}{2}\bar{m}_y^{\bar{\omega}_y}c_z^\beta\right)\beta=0. \qquad (19.5)$$

If disturbance/perturbation is caused by deflection of rudder then in right side of equation appears term $\mu\bar{m}_y^{\delta_н}\delta_н$. In this equation:

β - slip angle;

$$\bar{t}=t:\tau,\ \tau=\frac{m}{\rho SV};$$

$\bar{m}_y^{\bar{\omega}_y}=\frac{\partial m_y}{\partial\bar{\omega}_y}\frac{1}{\bar{r}_y^2}$ - derivative, which characterizes oscillation damping of aircraft relative to axis/axle Oy_1;

$\bar{\omega}_y=\frac{\omega_y l}{2V}$ - dimensionless angular yaw rate;

l - spread/scope; $\bar{r}_y^2=\frac{4I_y}{ml^2}$ - dimensionless radius of inertia relative to axis/axle Oy_1 of aircraft (squared);

$$c_z^\beta=\frac{\partial c_z}{\partial\beta};\quad \mu=\frac{2m}{\rho Sl};$$

$\bar{m}_y^\beta=\frac{\partial m_y}{\partial\beta}\frac{1}{\bar{r}_y^2}$ - derivative, which characterizes the degree of weathercock stability;

$\bar{m}_y^{\delta_н}=\frac{\partial m_y}{\partial\delta_н}\frac{1}{\bar{r}_y^2}$ - derivative, which characterizes the yaw moment effectiveness.

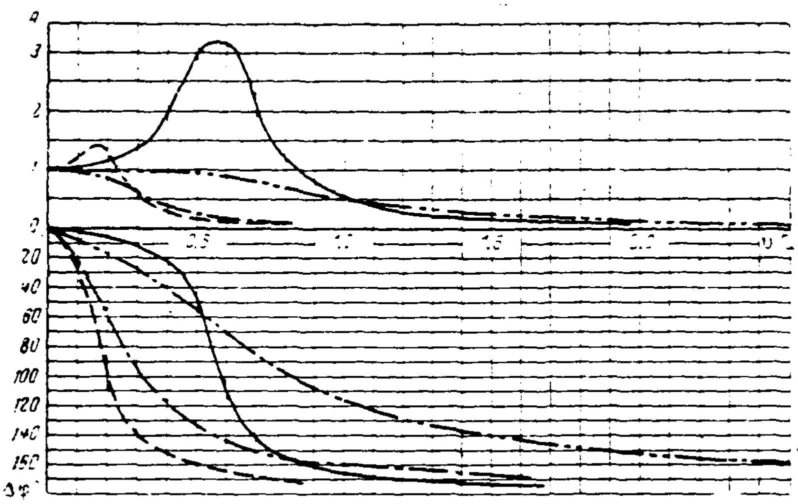

Fig. 19.4. Amplitude-phase frequency characteristics of heavy and light aircraft in different flight conditions: $\bar{A}=A/A_0$ (where A - amplitude of change in angle of attack with periodic control displacements with changing frequency; A_0 - change of angle of attack under the static conditions ($\nu=0$) with deviation of the same control); $\Delta\varphi$ - phase delay of change in angle of attack with respect to change in angle of deflection of altitude control; ν - frequency of alternating deviations of altitude control; ν_{co6} - natural frequency of aircraft.

Flight conditions.

The heavy nonmaneuverable aircraft: ——— - high altitude, M>1 (ν_{co6}=[illegible] Hz); ----- - high altitude, M<1 ($\nu_{co6}=0.195$ Hz); -··- low altitude, M<1, regime q_{max} ($\nu_{co6}=0.29$ Hz).

Light maneuverable aircraft: -•-•- low altitude, M<1, the regime of large velocity head (q) $\omega_{coc}=1.73$ Hz).

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In equation (19.5) analogous with equation (19.1) coefficient with first-order derivative characterizes damping natural oscillations (damping decrement).

For contemporary aircraft, especially during flights at high altitudes with high supersonic speeds, characteristically considerable decrease of damping. The corollary of this analogous with axial motion is the strong oscillation property of the processes of the lateral disturbed motion with the underdamping. Here one should recall that examined above equation (19.5), which describes the isolated/insulated motion of yaw, have given we only as the simplified example for the development/detection of the connections between the aerodynamic and inertial characteristics of aircraft. In reality the yawing motion of aircraft represents the set of the simultaneously developing rolls and yaw, in which the onset of slip causes the appearance of the moments of roll, and banking - yawing moments. Mathematically the connection/communication of the rolls and yaw in the yawing motion is illustrated by the known aerodynamic derivatives: m_x^{β}, $m_x^{\omega_y}$, $m_y^{\omega_x}$, m_{y}^{β} etc., and also by index $\varkappa=\bar{\omega}_{x\,max}/\bar{\omega}_{y\,max}$. In this case as a result of of the characteristic to the contemporary aircraft aerodynamic and inertial special features (large transverse static stability m_x^β at the high angles of attack and the low values of the moment of inertia I_x in comparison with I_y) the oscillatory roll is developed very intensely, so that index κ reaches values of 10 and more, and the oscillatory period becomes sufficient to small. Therefore in some flight conditions the analogous with pitch control inherent characteristics of pilot and in the duct/contour of lateral control can prove to be unsatisfactory for guaranteeing the stabilization of aircraft. Because of this control loop it will cease to be stable, the manifestation of what also proves to be the "rocking" of aircraft by pilot in the yawing motion.

At present stability of control loop is ensured by introduction to system of control of aircraft of special compensators, which work in parallel with pilot independent of it (vibration dampers, automatic machines of gear reduction, automatic feed units, etc.). These devices/equipment significantly change the dynamic characteristics of the aircraft (see, for example, Fig. 19.3a), improve handlings, raise flight safety and, is made itself piloting contemporary high-speed aircraft available for the pilots of average/mean qualification.

§3. Controls of contemporary aircraft.

On subsonic aircraft for longitudinal, transverse and azimuth guidance are used elevators, ailerons and rudders (Fig. 19.5). On the supersonic aircraft completely rotatable (or controllable) stabilizer is used, since elevator cannot ensure the necessary maneuverability at the high supersonic speeds as a result of the loss of efficiency (Fig. 19.6), while the longitudinal static stability of aircraft with respect to the g-force in these regimes significantly increases in connection with the bias/displacement of focus back/ago.

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For lateral control together with usual ailerons were adopted interceptors/spoilers (or spoilers), root ailerons, wing-tip ailerons - rotary wing tips and even differentially deviated right and left halves of stabilizer. The latter/last means of lateral control most frequently is used in the aircraft with the variable sweep wing.

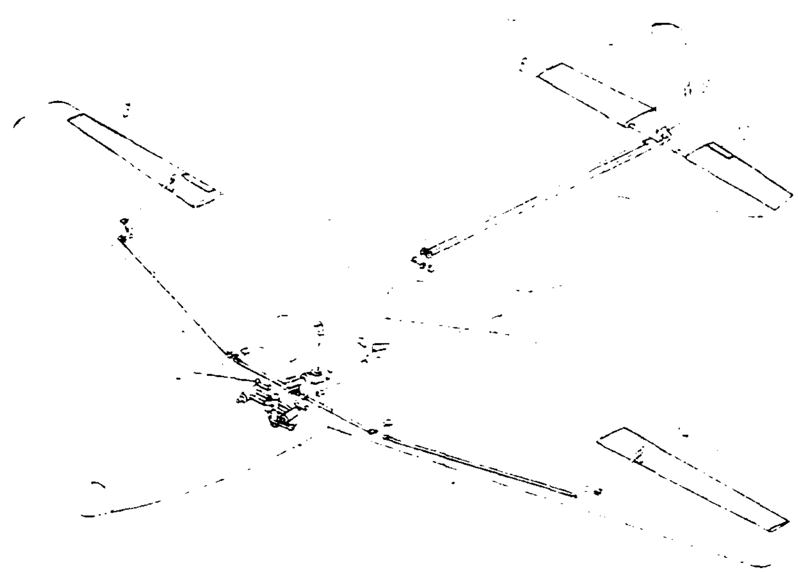

Fig. 19.5. Controls and control line of light low-speed aircraft: 1 - thrust of control line of ailerons; 2 - control pedal of rudder; 3 - aileron (right); 4 - trimer of aileron; 5 - control stick of elevator and of ailerons; 6 - elevator; 7 - fin; 8 - rudder; 9 - thrust of control line by rudder; 13 - control rod by elevator; 14 - aileron (left).

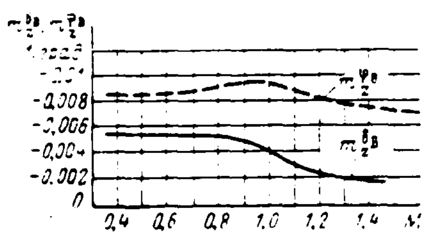

Fig. 19.6.

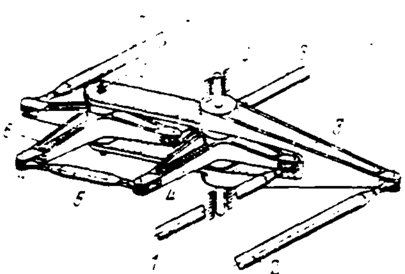

Fig. 19.7.

Fig. 19.6. Example of change in elevator-effectiveness derivative $(m_z^{\delta_B})$ and completely rotatable (controllable) stabilizer (m_z^{φ}) in dependence on flight Mach numbers.

Fig. 19.7. One of possible versions of kinematic mixing mechanism of control of elevons: 1 - control rod along bank; 2 - control rod on pitch; 3 - rocker-moving support; 4 - double-armed rocker of parallelogram mechanism of roll control; 5 - thrust of parallelogram mechanism; 6 - three-leg rocker; 7 - thrust to right elevon; 8 - thrust to left elevon; 9 - fastening rocker-moving support of mixing mechanism to construction of aircraft; 10 - rotational axis of three-leg rocker.

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For supersonic aircraft characteristically sufficiently rapid application of diagram "bobtailed aircraft", and sometimes also diagram "canard". The longitudinal and lateral controls of the aircraft of these diagrams are accomplished/realized by the elevons established/installed along the trailing wing edge, which deviate on right and left wings to one side for the execution of the functions of elevators or to the different sides - for accomplishing the functions of ailerons. For guaranteeing this in the control line to the elevons is established/installed the special mixing mechanism (Fig. 19.7), which summarizes command displacements over the elevator control (pitch) and by ailerons (bank) and puts out on elevons into one wing sum, and into another - a difference in these displacements.

Some of enumerated above new organs/controls of aerodynamic guidance are shown in Fig. 19.8 by conditionally combined on one hypothetical aircraft.

Let us pause now at special features of controls of vertically taking off (VTOL aircraft) and aerospace (VKS) aircraft.

In contrast to usual aircraft, which in process of takeoff/run-up for takeoff gain speed not only for creation of sufficient aerodynamic lift, but also for guaranteeing necessary efficiency of controls of aerodynamic guidance (controls), for VTOL aircraft and VKS are characteristic flight conditions, during which usual aerodynamic controls are not effective.

For VTOL aircraft - these are regimes of vertical takeoff and landing with in practice zero horizontal speed, at which usual aerodynamic controls become useless, although they completely effectively are used on these aircraft in other flight conditions with sufficiently high speed.

For VKS - these are flight conditions in rarefied layers of atmosphere, where shortcoming in velocity head also does not make it possible to use usual aerodynamic controls, successfully utilized under conditions of flight in denser layers of atmosphere.

Therefore for these aircraft together with usual organs/controls of aerodynamic guidance for guaranteeing possibility of changing position of aircraft and stabilization under conditions of vertical takeoff and landing (for VTOL aircraft) or superaerodynamic flight (for VKS) it is necessary to use other methods of designing of control forces and moments/torques. From these methods the widest use received:

1) air jet nozzle control;

2) a differential change (modulation) in the thrust of lift engines;

3) a change in the sense of the vector of the engine thrust.

During air jet nozzle control for creation of control forces and moments with respect to all three axes/axles of aircraft reacting forces of jets, produced behind special nozzles, arranged/located at maximally possible removal/distance from center of gravity of aircraft, are used.

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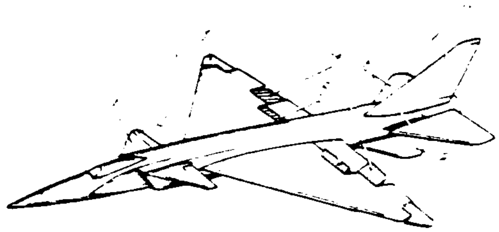

Fig. 19.8. New controls of aircraft (transverse controls are conditionally shown in the position for creating a left bank): 1 - horizontal nose empennage (destabilizer); 2 - flap of nose empennage; 3 - wing - tip aileron (right); 4 - wing-root ailerons; 5 - interceptors (spoilers); 6 - differentially deflected right and left halves of the stabilizer (ailerons); 7 - full-turn fin; 8 - lower (below fuselage) fin.

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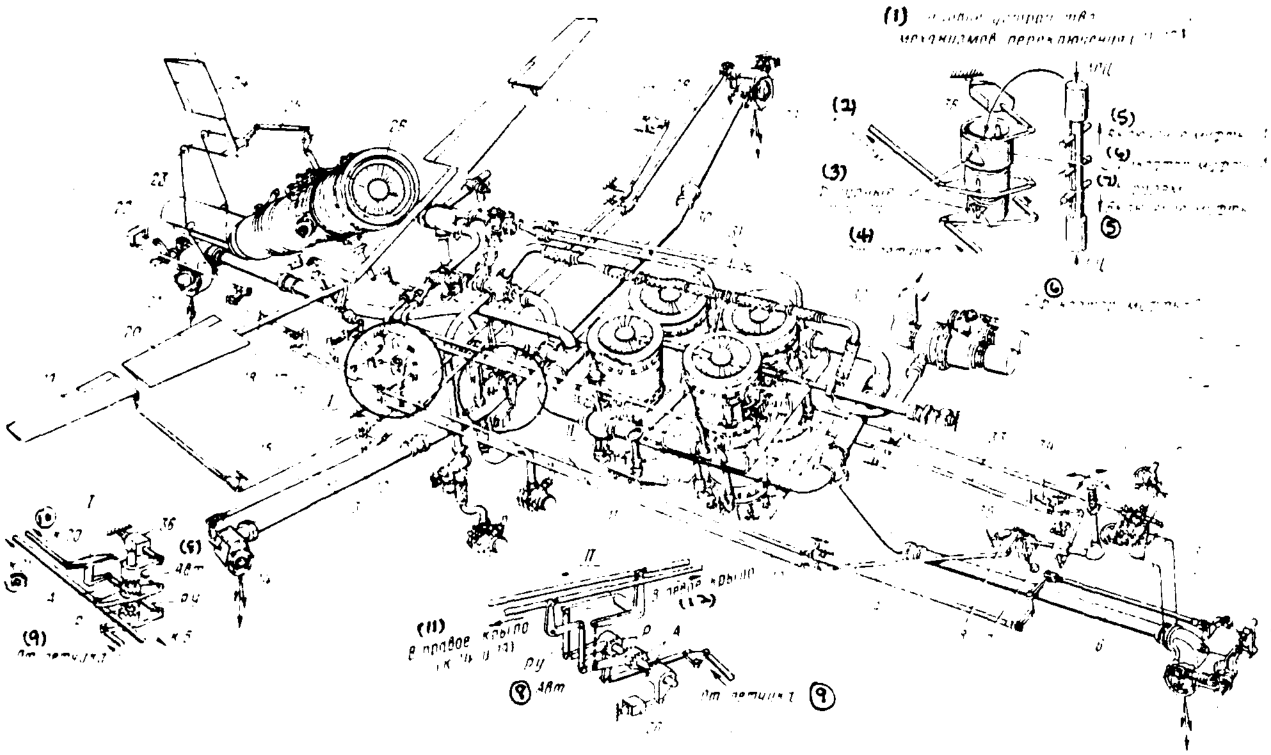

Fig. 19.9. Schematic of control of experimental vertically taking off aircraft Shortt SC-1 (England): 1 - control stick along pitch and bank; 2 - control pedal on yaw; 3 - the cable run of the rotation of front/leading nozzle for the yaw steering; 4 - mechanism of control of the valve of forepart/nose nozzle; 5 - forepart/nose nozzle of control with respect to pitch and yaw; 6 - air duct to the forepart/nose nozzle; 7 - potentiometric position detectors of knob/stick during pitch control (for the system of autostabilization); 8 - mechanism of the load of knob/stick during

pitch control; 9 - control line by the valve of forepart/nose nozzle; 10 - control line on the pitch (from the knob/stick); 11 - mechanism of the mode switch of roll control; 12 - air duct to the right nozzle of roll control; 13 - control line of the valve of right nozzle; 14 - right nozzle of roll control; 15 - control line of right aileron; 16 - mechanism of the mode switch of pitch control; 17 - hydraulic booster of elevator drive; 18 - air duct to the tail nozzle; 19 - aileron (right); 20 - elevator (right section); 21 - tail nozzle of control with respect to pitch and yaw; 22 - control line of the valve of tail nozzle; 23 - the cable run of the rotation of rear nozzle for the yaw steering; 24 - rudder; 25 - control line of rudder; 26 - sustainer TRD; 27 - control line of left aileron; 28 - control line of the valve of the left nozzle of roll control; 29 - left nozzle of roll control; 30 - air duct to the left nozzle; 31 - lifting TRD (4 pcs); 32 - air collector/receptacle; 33 - wiring/run roll control (from the knob/stick); 34 - mechanism of the load of knob/stick during the roll control; 35 - potentiometric position detectors of knob/stick during the roll control (for the system of self-stabilization); 36 - drives of the system of self-stabilization; I, II - mechanisms of the mode switch of pitch control and bank respectively.

Key: (1). Standard device of switching mechanisms (11.16). (2). To nozzles. (3). Pattern slots. (4). From pilot. (5). Inclusion of coupling. (6). Fixing arm of coupling. (7). To controls. (8). Automatic. (9). From pilot. (10). To. (11). To right wing (to 14 and 19). (12). To left wing.

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For air jet nozzle control on VTOL aircraft air, selected/taken from engines, which create vertical thrust (hoisting or lifting-sustainer), usually is used. Air exhaust through one or the other nozzle is accomplished/realized by an opening of the valve, connected with wiring/run with the appropriate control lever of aircraft in the cabin/compartment (by knob/stick, by handwheel or by pedals) and with usual control surface. Thus, during the motions of control lever simultaneously occurs the deflection of control and the opening of the corresponding valve of the system of air jet nozzle control.

Fig. 19.9 as example of this system shows system of jet-edge and aerodynamic guidance of English experimental VTOL aircraft Short SC-1. In this aircraft three control modes are possible:

1 - manual control (from the pilot) of nozzles and of the controls (rods in mechanisms 11 and 16 are established/installed in position RU or Avt, are included clutches P);

2 - manual control of controls, automatic control of nozzles from the system of the self-stabilization of thrust in the mechanisms are established/installed to the position RU, they are included/switched on clutch A);

3 - automatic control of stabilization with the help of the controls and the nozzles (rods in the mechanisms - in the position Avt., are included/switched on clutch A).

Switching clutches is accomplished/realized by hydraulic cylinder (GTs), and in emergency case (failure of hydraulics) clutches P are switched on by emergency pneumatic cylinder (APTs), which provides possibility of manual control of controls and of nozzles. Thrusts in the mechanisms of the mode switch of control of 11 and 16 are set in positions RU or Avt. on the earth/ground before the flight.

Air bleed from engines for system of air jet nozzle control leads to thrust decay of these engines, which strongly depends on relative quantity of selected/taken air $\bar{G}_{отб}=G_{отб}/G_{дв}$. As can be seen from Fig. 14.10, this thrust decay can comprise to 15-35% of initial engine thrust.

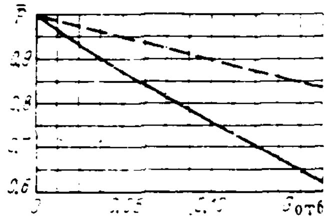

Fig. 19.10.

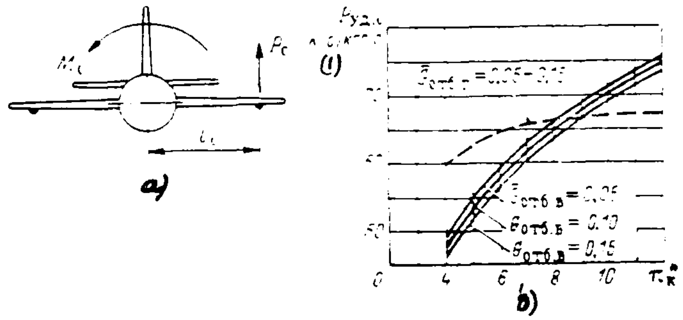

Fig. 19.11.

Fig. 19.10. Relative decrease of thrust of lifting TRD with increase in air bleed after compressor (——) or gases behind turbine (----)

Fig. 19.11. To determination of required air flow rate through steering jet: a) example of determination of arm of reaction force of nozzle relative to axis/axle of aircraft (center of gravity); ---- - gas bleed behind turbine; —— air bleed after compressor; 6 - dependence of specific thrusts of managers of nozzles, which operate on compressed of air from compressors and on exhaust gases of TRD, on compression ratio of air in compressors of engines.

Key: (1). kg/kg•s.

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One should note that thrust decay is significantly less, if we accomplish/realize selection not of air after the compressor, but gases behind the turbine. But with the latter/last method strongly is complicated entire system of air jet nozzle control, which must work on the high-temperature gas mixture.

Required control forces from nozzles of jet (jet-edge) control with known sizes/dimensions of aircraft, its weight and inertial characteristics can be determined from relationship/ratio (Fig. 19.11a)

$$P_{\text{потр.с}}=\frac{J}{l_{\text{с}}}\varepsilon_{\text{потр}}$$

where $P_{\text{потр.с}}$ - required control force of nozzle relative to axis/axle of aircraft in question;

J - moment of inertia of aircraft relative to this axis/axle;

$l_{\text{с}}$ - distance from axis/axle to nozzle;

$\varepsilon_{\text{потр}}$ - required during control angular acceleration of aircraft relative to the same axis/axle.

Value $\varepsilon_{\text{потр}}$ depends on weight category of aircraft and its aerodynamic characteristics and is chosen either by processing statistical data or according to results of simulation of processes of motion of VTOL aircraft on electron analogues.

After calculation $P_{\text{потр.с}}$ necessary maximum air flow rate through nozzle $G_{\text{р.с}}$ is determined from expression

$$G_{\text{р.с}}=P_{\text{потр.с}}\;P_{[illegible]\text{.с}}\ \text{kgf/s},$$

where $P_{уд}$ - specific thrust of nozzle (kg) with air flow rate through it in 1 kgf/s, it is removed/taken from graph Fig. 19.11b for each concrete/specific compression ratio of air in compressors of selected type of lift engines $\pi_{к}^{*}$ and relative quantity of selected/taken air or gases $\bar{G}_{отб}$. The required air flow rate through the engines in this case is counted according to expression $G_{в.в}=\frac{G_{в.с}}{\bar{G}_{отб}}$ kgf/s.

For air jet nozzle control on VKS instead of air are used decomposition products of any chemical substances, for example, peroxide of hydrogen (Fig. 19.12) [1].

FOOTNOTE [1]. In more detail about use and calculation of air jet nozzle control see, for example, [19], whence the given above design characteristics are used. ENDFOOTNOTE.

On VTOL aircraft with several-lifting and lift-sustainer-engines, spread along the length of fuselage and on wingspan, for positioning of aircraft and stabilization under conditions of vertical takeoff and landing can be used differential change in thrust of diverse engines or change in sense of the vector of thrust via rotation of entire engine or its nozzle cascade.

Example of layout of assemblies of control of this aircraft is shown in Fig. 19.13. Here at the low speeds of flight pitch control and bank is accomplished/realized by a differential change in the thrust of diverse along the fuselage and on the span of the wing of groups lifting and lift-sustainer-engines, and yaw steering - by thrust rotation (by rotation of the wing pods of lift-sustainer-engines). For the change the senses of the vector of the thrust of lifting-sustainer TRD in the transient regimes of nacelle simultaneously are turned from knob/stick 3 at angle of 90°, while for the yaw steering nacelles are turned from pedals 1 to the small angles to the different sides.

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Control with the help of change in thrust of lifting TRD as a result of insufficiently high accelerating of latter possesses somewhat worse/worst characteristics of operating speed, and it also requires decrease of general/common vertical thrust with its differential change in diverse engines for creation of manager of moment/torque.

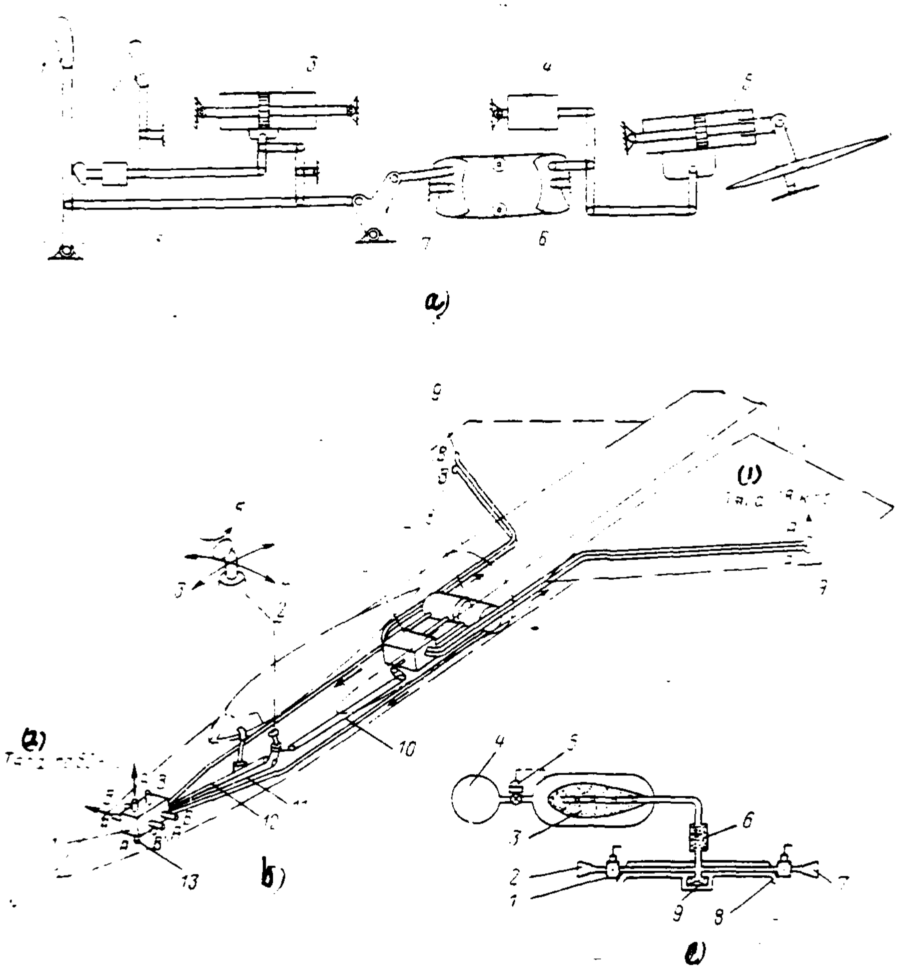

Fig. 19.12. System of control of experimental aircraft-boost-glide vehicle of North American X-15 (USA): a) diagram of longitudinal aerodynamic guidance: 1 - central (fundamental) control stick; 2 - lateral knob/stick (on right panel); 3 - auxiliary hydraulic booster; 4 - actuating mechanism of the damper of longitudinal oscillations (pitch); 5 - hydraulic booster of the drive of the surface of pitch

control (completely rotatable stabilizer); 6 - cable run; 7 - mechanism of nonlinear transfer; 8 - automatic feed unit; b) the schematic of the arrangement/position of the assemblies of the jet (jet-edge) system for the control: 1 - duplicated/backed up control valves with respect to pitch and yaw; 2 - control stick of jet system; 3, 4, 5 - control of knob/stick according to pitch, bank and yaw respectively; 6 - duplicated/backed up control valves along the bank; 7, 8 - hydrogen peroxide tanks; 9 - duplicated/backed up stability-guidance jets along the bank; 10, 11, 12 - cable control line by the valves of bank, pitch and yaw respectively; 13 - stability-guidance jet with respect to pitch and yaw in the forward fuselage; A - fundamental nozzles with the feed/supply by peroxide from tank 7; B - duplicating nozzles with the feed/supply from tank 8; c) the schematic diagram of the reaction control system, which works on hydrogen peroxide: 1 - control valve (opening output to the gases through the nozzle); 2 - nozzle; 3 - hydrogen peroxide tank; 4 - tank/balloon with nitrogen; 5 - pressure regulator; 6 - gas generator; 7 - second nozzle of system; 8 - drainage conclusions/outputs; 9 - safety valve.
Key: (1). Thrust 18 kg. (2). Thrust on 50 kg.

S4. Aircraft handling, the most important characteristics.

Briefly let us pause at aircraft handling and its characteristics [1].

FOOTNOTE [1]. In more detail this question can be studied, for example, according to book [11]. ENDFOOTNOTE.

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By aircraft handling is understood its ability to change flight conditions as a result of actions of pilot, which are reduced to displacements of control levers during application to them of specific efforts/forces. Therefore controllability is characterized by the dependences between the deviations of control levers and the efforts/forces applicable to them, on one hand, and by changes in the parameters of the motion of aircraft - on the other. In the practice frequently of handling they divide into two groups: static and dynamic controllability. Although this distribution and sufficiently conditionally, it nevertheless extensively is used.

Characteristics of static controllability determine force feel and their deviations, necessary for balancing/trimming of aircraft in different flight equilibria, and also change in these efforts/forces and displacements, necessary for transition/transfer from one trimmed/steady-state regime to another with unit change in parameters of motion. Most important of these characteristics in the pitch control are:

1) balancing curves $x_{в}=f_1(M, H, n_y)$; $P_{в}=f_2(M, H, n_y)$ and, in the particular case, balancing curves of level flight $x_{в\,г\,п}=f_3(M, H, \bar{x}_T)$ and $P_{в\,г\,п}=f_4(M, H, \bar{x}_T)$.

Conventional designations here and throughout are used: δ - deflection of control; x - linear displacement of control lever (usually in practice it is measured in mm); P - force feel. Indices "в", "э", "н" designate respectively the channels of the elevator control (pitch), ailerons (bank) and rudder (yaw);

2) the expenditures of the lever of pitch control and efforts/forces on it for a unit change in g-force $x_{в}^{n_y}=\partial x_{в}/\partial n_y$ and $P_{в}^{n_y}=\partial P_{в}/\partial n_y$ or speed $x_{в}^{V}=\partial x_{в}/\partial V$ and $P_{в}^{V}=\partial P_{в}/\partial V$;

3) the expenditures of efforts/forces per the lever of pitch control during the recovery from the level flight to the flight conditions with the maximum permissible values of coefficient $c_{y\,доп}$ or the maximum sizes of the g-force $(P_{в})_{c_{y\,доп}}$ and $(P_{в})_{n_{y\,доп}}$ respectively.

As characteristics of lateral static controllability are considered dependences of deviations of lever of lateral control (knob/stick or handwheel) $x_{э}$ and biases/displacements of pedals $x_{н}$ and also appearing on them efforts/forces $P_{э}$ and $P_{н}$ respectively from parameters of yawing motion of aircraft, as which are examined angular velocities ω_x and ω_y, angles of slip β and of bank γ, and also lateral acceleration n_z. For the evaluation of lateral static controllability also they are used:

1) balancing curves $x_э=f_1(\beta \text{ or } n_z)$, $P_э=f_2(\beta \text{ or } n_z)$, $x_н=f_3(\beta \text{ or } n_z)$ and $P_н=f_4(\beta \text{ or } n_z)$, determined from a series of rectilinear flights with slipping, and also in steady flights with unsymmetrical thrust (for the multiengine aircraft);

2) dependences $\omega_x=f_5(x_э \text{ or } P_э)$, including $\omega_{x\,max}=f_6(x_{э\,max} \text{ or } P_{э\,max})$ in different flight conditions;

3) derivatives of displacements and efforts/forces on the levers of transverse and azimuth guidance on any to the parameter of the yawing motion, undertaken under the conditions of the initial steady rectilinear laterally level flight and slip

$x_э^{\omega_x}=\partial x_э/\partial\omega_x$, $P_э^{\omega_x}=\partial P_э/\partial\omega_x$, $P_э^{\gamma}=\partial P_э/\partial\gamma$, $P_н^{\gamma}=\partial P_н/\partial\gamma$, $x_э^{\beta}=\partial x_э/\partial\beta$, $P_э^{\beta}=\partial P_э/\partial\beta$

etc.

For contemporary high-speed aircraft sufficiently important prove to be also balancing curves, which characterize dependences of efforts/forces on levers of transverse and azimuth guidance and their displacements on speed (or Mach number) of straight flight with $\beta=\gamma=0$.

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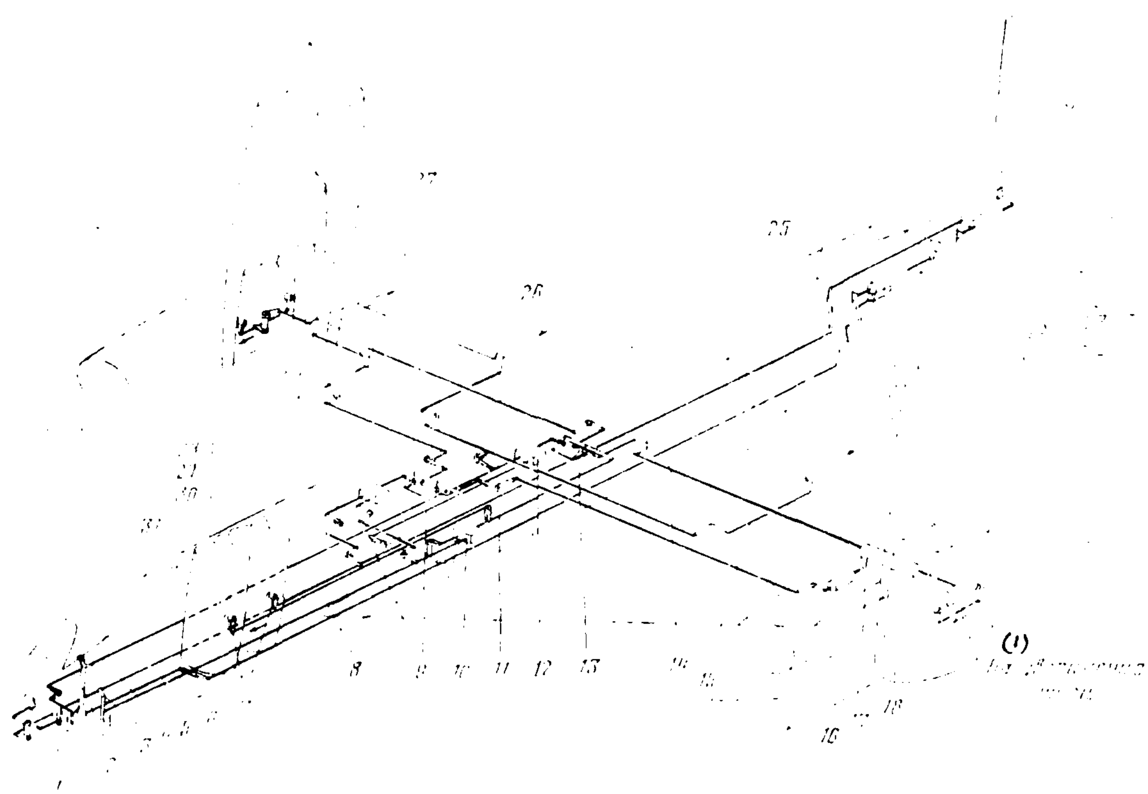

Fig. 19.13. Schematic diagram of mechanical feature of system of control of experimental VTOL aircraft VJ-101C (FRG): 1 - control pedal on yaw; 2 - control stick along the pitch and the bank; 3 - control stick of the rotation of nacelles for a thrust deviation during the transient regimes (from the vertical takeoff to the dispersal/acceleration and from the level flight to the vertical landing); 4 - control line along the bank; 5 - control line on the

pitch; 6 - control line of nacelles; 7 - control levers of thrust of TRD (in shown the direction of the motion of levers to thrust augmentation); 8 - control line of the thrust of fuselage group; 9 - wiring/run to the mechanisms of the cutoff/disconnection of control of thrust upon the rotation of nacelles and transfer to the level flight; 10 – mechanism of the cutoff/disconnection of control of thrust according to pitch; 11 - control line of the thrust of the groups of engines in the nacelles of pitch control; 12 - mechanism of disconnection of control of bank thrust; 13 - mixing mechanism of control of thrust on pitch and bank (kinematics of mechanism see in Fig. 19.7); 14 - control line in the wing of control of the thrust of the group of lifting-sustainer TRD in left nacelle; 15 - control line of left aileron; 16 - control line of the rotation of left nacelle; 17 - rotational axis of nacelle with the lever; 18 - left nacelle with two lifting-sustainer TRD (just as right 27, is arranged/located after the center of gravity of aircraft); 19 - aileron left; 20 - control line by completely rotatable stabilizer; 21 - hydraulic booster of stabilizer; 22 - rudder power jack; 23 - controllable stabilizer (left half); 24 - rudder; 25 - control line of rudder; 26 - aileron (right); 28 - mixing mechanism of control of the rotation of nacelles; 29 - mechanism of the cutoff/disconnection of the system of the rotation of nacelles for the yaw steering; 30 - the control line of the rotation of nacelles for control of yaw; 31 - fuselage group of lifting TRD (it is arranged/located in front of the center of gravity of aircraft).

Key: (1). For increasing thrust.

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These curves at different heights/altitudes characterize force feel and their displacements/movements, necessary for countering of the moments of roll and yaw, which appear as a result of the compressibility effect of air, structural elasticity of aircraft or its production asymmetry.

Dynamic controllability characterizes aircraft handling with execution of different unsteady motions, special case of which can be translation/conversion of aircraft of one flight equilibrium in another. We already mentioned, that any deviation of any control, the calling change in the acting on the aircraft forces and moments/torques, can be considered as the disturbance/perturbation, which deflects/deviates aircraft from the reference-flight conditions. The behavior of aircraft after this deviation is described by appropriate equations of motion, from which, as are more important, are examined the equation of the longitudinal and lateral short-period motions [see equations (19.1) (19.5) with the right sides]. The process of changing the parameters of aircraft upon transfer from one flight equilibrium to another after the control displacement (or the process of return to the reference-flight conditions after the action of external disturbance/perturbation) is called transient process, and for the characteristics of dynamic controllability accept the characteristics of this process (see Fig. 19.36).

Since transient-response characteristics are wholly defined by coefficients with corresponding derived parameters in equations of motion of aircraft, and its stability - by conditions of type (19.3), then it is possible to draw conclusion that characteristic of dynamic controllability (however, just as static) they are wholly determined by stability characteristics of aircraft. But these characteristics during the design of aircraft and its system for control can be changed by design engineers within sufficiently wide limits.

Values of most important indices of controllability for aircraft of different types are given on the basis of many-year practice of operation of aircraft. In the practice of the design of the systems of control of contemporary aircraft as a result of special research these indices can be corrected for obtaining their optimum values (the most convenient from the point of view pilots), and also for the stabilization of the control settigns "pilot - the system of control - aircraft" in all possible flight conditions.

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These research with the participation of pilots is conducted on the special stands with the use of analog computer technology (Fig. 19.14). Sometimes these stands for greatest approximation to the actual conditions for flight are made with the mobile cabins/compartments, which imitate the angular and even linear accelerations, which operate on the aircraft in flight. Control of the motions of this cabin/compartment is also realized by the electronic computers (computer(s)), which decide the equations of motion of aircraft.

§ 5. General design of aircraft and handling.

It was above noted that stability characteristics and aircraft handling can be changed during its design within sufficiently wide limits. After considering, for example, expression for the coefficients with the derivatives in the equation of longitudinal short- period motion (19.1), it is possible to note that such derived as $m_z^{c_y}$ and $m_z^{\delta_B}$ about the design of aircraft they can sufficiently easily be changed.

Derivative $m_z^{c_y}=\bar{x}_T-\bar{x}_F$, which characterizes to be changed longitudinal stability factor with respect to g-force, can be changed due to change in centering $\bar{x}_T$ and position of focus of aircraft $\bar{x}_F$. The position of focus, as is known, is determined by form, sizes/dimensions and wing characteristics and horizontal tail assembly, and also by their mutual location. Derivative $m_z^{\delta_B}=\partial m_z/\partial\delta_B$, which characterizes the elevator-effectiveness derivative, depends on form and sizes/dimensions of horizontal tail assembly and elevator and is determined by the characteristics of its profiles/airfoils, by the arm of horizontal tail assembly $L_{\Gamma.O.}$

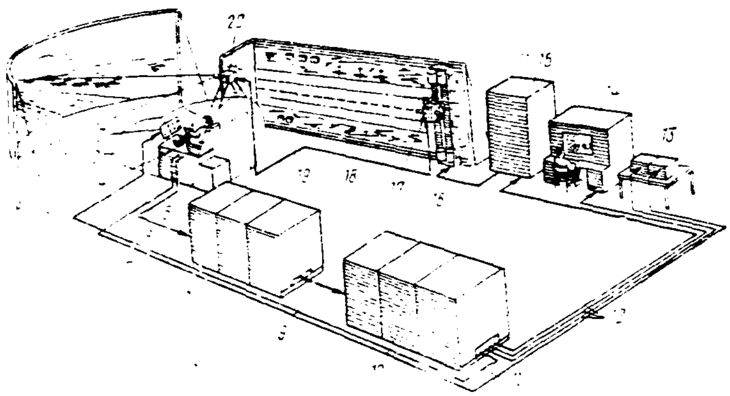

Fig. 19.14. Standard stand for research of dynamics of aircraft (with fixed cabin/compartment): 1 - screen/shield with mobile image of "terrain"; 2 - mock-up of the pilot's cabin; 3 - instrument panel of pilot; 4 - test pilot; 5 - unit of the load of the control levers and sensors of efforts/forces and displacements/movements; 6 - signals of the displacements/movements of control levers to the calculator; 7 - signals of efforts/forces and displacements/movements to recording equipment; 8 - electron analogue of the system of control of aircraft; 9 - signals of the control displacement (controls) of aircraft; 10 - electron analogue of the dynamics of aircraft; 11 - signals of a change in the parameters of flight to the instruments of pilot; 12 - signals of a change in the parameters of flight to the recording equipment, the panel for operator and the controlling unit of the system of the displacements/movements of the receiver of the "survey/coverage of terrain"; 13 - recording equipment (oscillographs); 14 - panel for the operator of stand; 15 - controlling unit of the system of the displacement/movement of the receiver of the "survey/coverage of terrain"; 16 - signals of a change

in the position of receiver; 17 - mobile television head of the receiver of the "survey/coverage of terrain" with six motions (3 progressive/forward and 3 they are rotary); 18 - terrain mockup (the runway of airfield); 19 - picture signals of terrain to the projector; 20 - projector.

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All these values sufficiently easily can vary in the process of design. With the insufficient elevator-effectiveness derivative is realized the transition/transfer to the completely rotatable controllable stabilizer, whose efficiency substantially above is rated/estimated by derivative $m_z^{\varphi_B} = \partial m_z / \partial \varphi_B$ (cm. Fig. 19.6).

Very high value has correct; selection of value m_z^{φ}. In Chapter VI has already been indicated that an increase in this derivative leads to an increase in the losses of quality to the balancing/trimming. On a par with this, an increase in the stability level, especially with the decrease of the damping (precisely this change in these characteristics it occurs upon transfer to supersonic speeds), leads to the increase of initial overshoot (overcontrol, see Fig. 19.3b) the controlled parameter at the deflections of elevator. On the other hand, the decrease of longitudinal stability factor with respect to the g-force leads to the decrease of the most important indices of longitudinal static controllability $x_B^{n_y}$ and $P_H^{n_y}$. and their excessive decrease can lead to the loss of stability of control loop. The dependence of index $x_B^{n_y}$ on the longitudinal stability factor with range/trimming respect to the g-force takes the following form:

$$x_{B}^{n_y} = -\frac{1}{k_{ш.B}} \frac{c_y [illegible]}{m_z^{B}} \left(m_z^{c_y} - m_z^{[illegible]}\right), \qquad 19.6$$

where

$k_{ш.B} = \frac{d[illegible]}{dx_B}$ - coefficient of kinematic transfer in wiring/run of pitch control.

This expression easily can be obtained from equation (19.1) with right side, written for establishing the angle of attack $\left(\frac{d^2\Delta\alpha}{dt^2} = \frac{d\Delta\alpha}{dt} = 0\right)$ converted with the help of relationship/ratio (19.2) for determining increment in g-force. During the fully powered controls the index of controllability $P_B^{n_y}$ is determined by the simple expression

$$P_B^{n_y} = P_B^x x_B^{n_y}, \qquad 19.7$$

where coefficient $P_B^x = dP_B/dx_B$ - the gradient of load it growled pitch control according to its deviation, created by loader.

From expression (19.7) it follows that with the constant characteristic of load decrease of index $x_B^{n_y}$ with decrease $m_z^{c_y}$ leads also to appropriate decrease $P_B^{n_y}$. Thus, during the design of aircraft it is some method to satisfy the contradictory requirements: on one hand, attempts to do $m_z^{c_y}$ how it is possible less for decreasing the losses of quality to the balancing/trimming for the long-range aircraft or improvements in the maneuverability - for easy, and on the other hand, to ensure acceptable handlings.

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Expressions (19.6) and (19.7) to a certain extent suggest path for solution of this problem. As it follows from (19.6), even with sufficiently low value $m_z^{c_y}$ it is possible to ensure the acceptable values of index $x_{в}^{т}$ with three methods:

1) by decreasing the transmission factor from the lever of pitch control for the elevator (or stabilizer) $k_{ш.в}$.

2) by decreasing the pitch moment effectiveness $m_z^{\delta_в}$;

3) by an increase in longitudinal damping $m_z^{\bar{\omega}_z}$.

Values of index $P_{в}^{т}$, besides enumerated above three methods, can be led to acceptable level by increase in gradient of load $P_{в}^{x}$.

All these methods extensively are used in contemporary aircraft construction and is found their technical realization;

- in regulation of gear ratios on flight conditions;

in - application of combined type pitch control, with which as altitude control to some regimes (M<1) is used elevator, and on others (M<1) - controllable stabilizer;

- in increase in damping aircraft $m_z^{\bar{\omega}_z}$ due to installation of automatic devices/equipment - pitch dampers;

- in application of regulation of gradient of load $P_{в}^{x}$ on flight conditions.

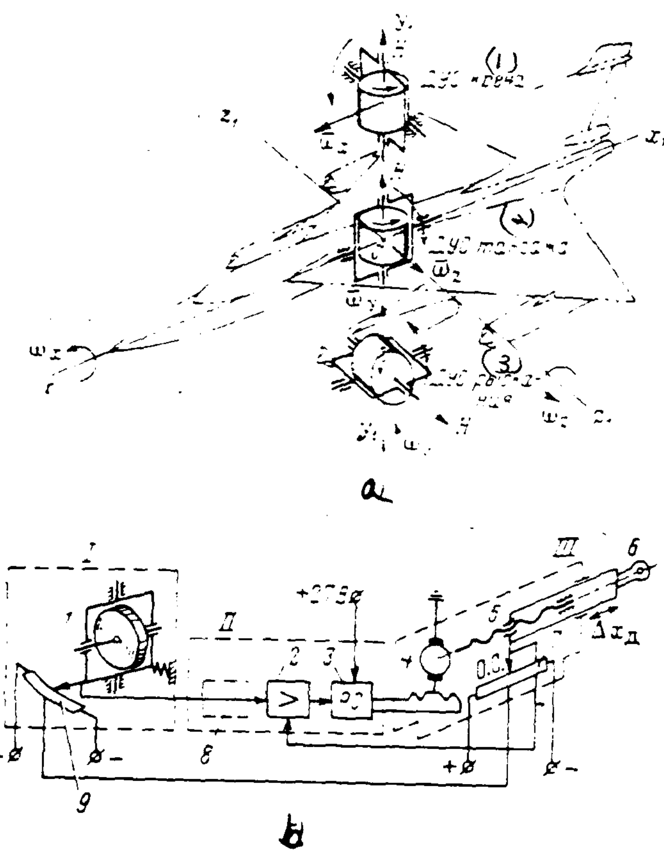

Fig. 19.15. The explanation of work of vibration dampers of the aircraft: a) diagram of installation of gyroscopes of sensors of angular velocities (DUS) of roll dampers, pitch and yaw (H - vector of moment of momentum of gyrorotor); b) schematic diagram of damper; 1 - gyroscope of sensor of angular velocity; 2 - amplifier; 3 - power relay (RS); 4 - electric motor; 5 - reducer-converter of rotation of electric motor in forward motion of stock/rod; 6 - output stock/rod; 7 - feedback potentiometer (OS); 8 - filter of high frequencies; 9 - potentiometer of sensor (by dotted line they are shown basic building blocks of assembly of damper): I - sensor of angular velocity (DUS);

II - relay-amplifier unit (RYB); III - actuating mechanism thrust.
Key: (1). bank. (2). pitch. (3). yaw.

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In each specific case depending on the aircraft type, its characteristics and presented requirements one or the other method or their combinations is used.

Practice shows that it is impossible significantly to change unfavorable dynamic characteristics of aircraft at high supersonic speeds on large altitudes only by methods of aerodynamic layout. This problem is solved by applying the automatic means of an increase in the damping - vibration dampers.

Depending on that, relative to which of axes of aircraft damper must extinguish vibration, pitch dampers, bank and yaw are distinguished. Their device/equipment is completely equal, and they are characterized by only the installation of their sensing elements - gyroscopes (Fig. 19.15a). Sensing element of damper (gyroscope) reacts to a change in the angular velocity of aircraft and puts out signal to the actuating mechanism - telescopic thrust (Fig. 19.15b). Telescopic thrust is established/installed consecutively/serially in the control line. Therefore with the movements of the stock/rod of telescopic thrust to values Δx_1 is introduced the disagreement/mismatch between the positions of the lever in the cabin/compartment and of control (control or the valve of jet-edge system on the VTOL aircraft). Therefore the control with the fixed control lever is deflected to the appropriate side for the countering of the rotation of aircraft.

During design of control system optimum laws of work and gear ratios of dampers [1], which ensure necessary damping of natural oscillations of aircraft, are chosen (by calculation or by modeling of processes on electron analogues), which ensure necessary improvement of characteristics of static controllability and stability of duct/contour "damper - control system - aircraft".

FOOTNOTE [1]. The number, which shows, is called the gear ratio of damper (k_{ω}) on how many degrees are deflected by damper control for the countering of the unit angular rate of rotation of aircraft; dimensionality grad/grad=s. ENDFOOTNOTE.

If pitch damper works according to the law $(\delta_в)_{д.т}=k_{\omega_z}\omega_z$ (gear ratio k_{ω_z}) then during its installation absolute value of derivative increases by $(\Delta m_z^{\omega_z})_{д.т}=k_{\omega_z} m_z^{\delta_в}\frac{V}{b_A}$, and index $x_в^n$ on $(\Delta x_в^n)_{д.т}=-\frac{k_{\omega_z}}{k_{ш.в}}\frac{g}{V}\cdot 57.3$ (Fig. 19.16). As can be seen from equation (19.1), an increase in the absolute value of derivative $m_z^{\omega_z}$ leads to an increase in the coefficient with dΔα/dt (since $m_z^{\omega_z}<0$) the determining degree of damping of the natural oscillations of aircraft, and, consequently, it leads to their more rapid attenuation.

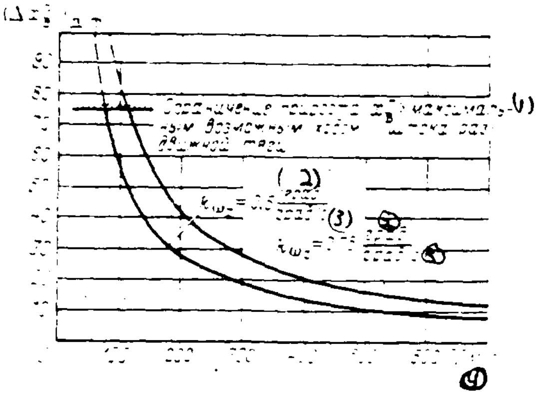

Fig. 19.16. Change on flight speed of increments in index of controllability x_B with work of pitch damper with different gear ratios (k_{ω_z}= deg/mm).

Key: (1). Limitation of increment ... by the maximum possible course of rod of telescopic pull. (2). deg/. (3). deg/s. (4). m/s.

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As shows study of equations of yawing motion [type (19.5)], greatest role in formation of characteristics of lateral controllability play following aerodynamic derivatives.

1) m_x^β - effective-dihedral derivative $m_x^\beta < 0$;

2) m_y^β - the degree of weathercock stability $m_y^\beta < 0$;

3) $m_y^{\omega_y}$ - the derivative, which characterizes oscillation damping of yaw (relative to axis Oy_1 of aircraft).

Value of derivative m_x^β varies with change in forms and sizes/dimensions of wing and vertical tail assembly, and also to a considerable extent with change in dihedral angle of wing. This one can see well from the approximation, which determines the value of derivative m_x^β:

$$m_x^\beta = m_{x\,\text{кр}}^\beta + m_{x\,\text{в.о}}^\beta = m_{x\,\text{кр}}^\beta - \frac{1}{2} c_y^\alpha \bar{z}_{\text{ст}} V \cos^2\chi - \frac{1}{2} \bar{z}_{\text{ст}} c_y \sin\chi - k a_{\text{в.о}} \frac{S_{\text{в}}}{S} \frac{y_{\text{в.о}}}{l}, \quad (19.8)$$

where χ - sweep angle of wing;

V - dihedral angle of wing;

$y_{\text{в.о}}$ - coordinate of the point of the application of lateral force on the vertical tail assembly relative to axis Ox_1 of aircraft;

$$a_{\text{в.о}} = \partial c_{z\,\text{в.о}} / \partial \beta;$$

$\bar{z}_{\text{ст}}$ - distance from the plane of the symmetry of aircraft to the center of gravity of the area of half wing, in reference to the semispan; for tapered wing $\bar{z}_{\text{ст}} = \frac{1}{3} \frac{\eta - 2}{\eta - 1}$ (η - wing taper);

l - wingspan.

As it follows from expression (19.8), on aircraft, whose fundamental parameters (S, χ, $S_{\text{в.о}}$, c_y^α) are already selected, for decreasing of absolute values m_x^β and improvement thereby of dynamics of yawing motion at high angles of attack it is simplest to use negative dihedral of wing (V<0).

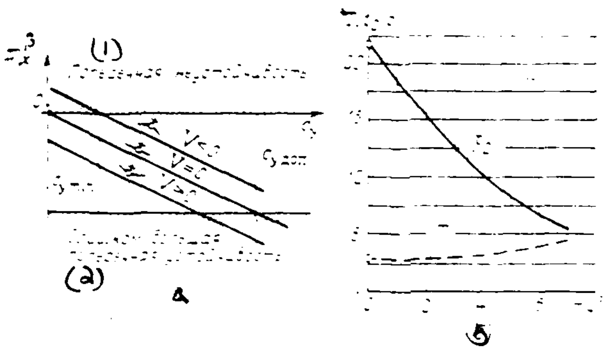

Fig. 19.17. Effect of change dihedral of wing to static and dynamic characteristics of aircraft in yawing motion: a) character of change in dependence of aircraft with sweptback wing; b) change in characteristics of lateral dynamic stability of aircraft Boeing B-47 (USA) under conditions of landing approach (H=0, V=240 km/h, flaps they are deflected); T - period of lateral short-period vibration; t_2 - time of decrease of amplitude of oscillations/vibrations doubly.
Key: (1). The lateral instability. (2). Too great lateral a stability.

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In this case this in expression (19.8) the second term becomes positive and the negative values of derivative m_x^β are reduced (Fig. 19.17).

However, as is evident, from Fig. 19.17a, at low angles of attack (flight on high speeds) this measure leads for excessive decrease of derivative m_x^β, reverse/inverse reaction along bank to deflection of rudder ("giving of leg") can appear.

At the same time, ends/leads fur-trimmed down of outer planes of wing with negative dihedral lead to whole series of difficulties with layout of chassis/landing gear of aircraft (it is necessary to increase length of struts), and also during design of fuel systems with fuel tanks mixed in wing, since fuel/propellant cannot of one's own accord/by gravity be supplied from outer planes of wing omitted down to service tanks.

Therefore at present problem of guarantee with accepatable dynamic characteristics of aircraft in yawing motion increasingly more frequently is solved only due to application of different automatic means (automatic machines of slip and directional- control dampers), and dihedral is used positvely. It is natural that in this case to the reliability of automatic systems the increased requirements are presented.

As far as derived m_y^β characterizing weathercock stability is concerned, this derivative in essence depends on relationship/ratio of projected areas on plane of symmetry of aircraft of forward fuselage and vertical tail assembly:

$$m_y^\beta \approx -0.06 c_y^\alpha \,\mathrm{tg}\,\chi - a\, m_y^\beta - k a_{в.} A_{в.} + k_ф \frac{h_ф l_ф^2}{S l}. \qquad (19.9)$$

where

$$k_ф \approx 0.32\, \bar{x}_ф - 0.25 + \frac{[illegible]}{\lambda_ф^{[illegible]}};\quad \bar{x}_ф = \frac{x_ф}{l_ф};\quad \lambda_ф = \frac{h_ф}{l_ф}.$$

The values of the entering expressions (19.8) and (19.9) designations of the geometric parameters of aircraft are given in Fig. 19.18.

Characteristic increase in lengths of fuselages $l_ф$ and their nose sections $x_ф$ in contemporary high-speed aircraft, connected with special features of their layout and need for maximum lowering of wave impedance, leads to increase of "destabilizing" latter/last term in expression (19.9) and, consequently, in to decrease of derivative m_y^β. At the same time, expression (19.9) shows that with an increase in the angles of attack (landing approach or flight at the high altitudes) the directional stability is also reduced both due to the unsatisfactory effect $(m_x^\beta)_{kr}<0$. and due to an increase in shading/blanketing vertical tail assembly by the fuselage, as a result of which is reduced the coefficient of braking flow in the zone of tail assembly $k=V^2_{B.O}/V^2<1$.

With large Mach numbers derivative m_y^β is reduced as a result of reduction in efficiency of vertical tail assembly $(a_{B.O}=\partial c_{zB.O}/\partial\beta$ — decreases.

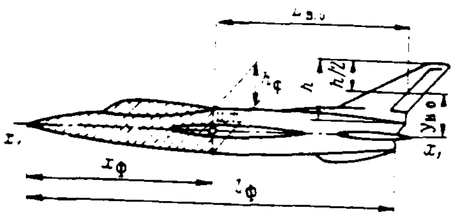

Fig. 19.18. Determination of geometric parameters of aircraft, entering expressions for derivatives m_y^β and m_x^β which characterize lateral static stability.

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All this leads to need for corresponding increase in fin-and-rudder area for guaranteeing directional stability, especially under conditions for supersonic flight at high altitudes (Fig. 19.19a).

Solution of questions of general/common design of high-speed aircraft is impossible also without guarantee of necessary degree of damping transverse oscillations, determined in by primary meanings of derivative $m_y^{\omega_y}$.

From approximation for this derivative

$$m_y^{\omega_y} \approx m_{y\,\text{в.о}}^{\omega_y} = -2.4 k a_{\text{в.о}} \frac{S_{\text{в.о}}}{S} \frac{L^2_{\text{в.о}}}{l^2} \qquad (19.10)$$

it follows that it sufficiently rapidly is reduced at high supersonic speeds (as a result of decrease $a_{\text{в.о}}$) and at high angles of attack (as a result of braking of flow and decrease k), and increase in vertical-tail length and its areas are most effective methods of its increase in selected parameters of wing.

However, as show calculations, for obtaining necessary values of this derivative is required either very considerable increase in area of tail assembly (Fig. 19.19b and c), which leads to considerable gain in weight of construction of aircraft, or sufficiently great increase in length of aft fuselage section L_{3}, leading to essential layout difficulties and furthermore to considerable gain in weight of construction/design.

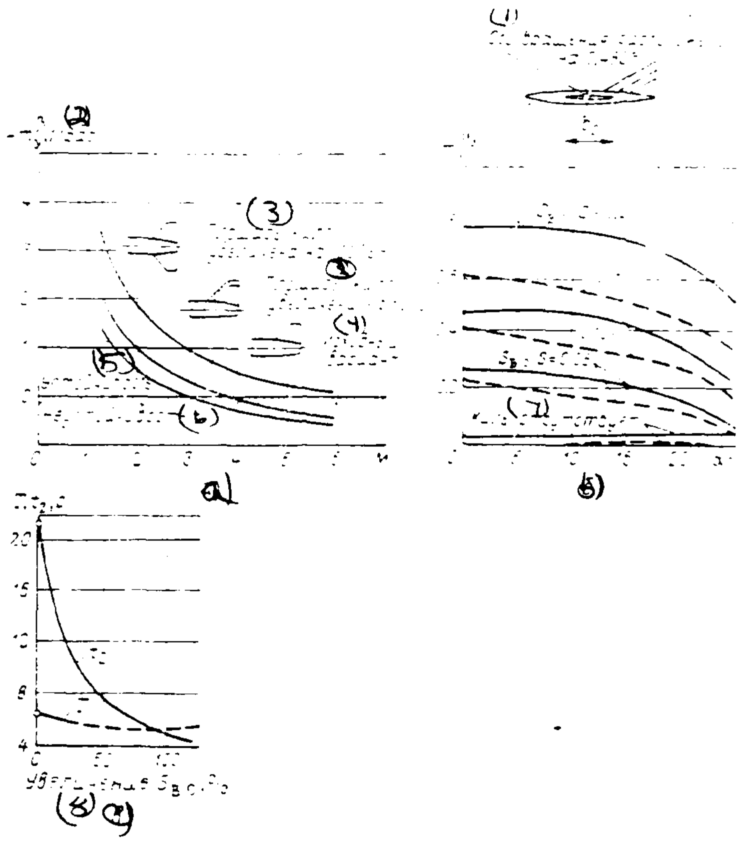

Fig. 19.19. Effect of change in fin-and-rudder area to static and dynamic lateral-behavior characteristics of aircraft: a) change in weathercock stability of aircraft at supersonic speeds; b) change in characterizing damping oscillations of yaw derivative $m_y^{\omega_y}$ at different angles of attack - --- averaged data of tests of firm. Bristol; - - - averaged data of tests RAE); c) a change of characteristics of the lateral dynamic stability of aircraft Boeing B-47 under conditions of landing approach (H=0, V=240 km/h, flaps they are deflected, the beginning of the coordinate axes, just as in Fig. 19.17b, it corresponds to the initial version of aircraft).

Key: (1). Rotational axis is arranged/located on. (2). rad. (3).

Area of fin increased by 50%. (4). Initial version. (5). Stability. (6). Instability. (7). Fin is absent. (8). Increase.

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Therefore installation into system of control of directional-control damper is most rational method of improvement in characteristics of lateral dynamic stability and controllability of contemporary aircraft. If directional-control damper works according to the law then increase of derivative during its installation can be evaluated with the help of the simple expression.

From this expression it follows that with appropriate increase in gear ratio of directional-control damper it is possible to obtain necessary increment in derivative . However, one ought not to forget that an increase in the gear ratio of damper is limited by the stability conditions of duct/contour "damper - the system of control - aircraft" [1], but the necessary increment in derivative can be provided only during the guarantee of the corresponding rudder-effectiveness derivative

FOOTNOTE [1]. Practice shows that for the contemporary heavy aircraft value $(k_{\omega_y})_{max}$ is limited approximately with the values of 1.5-2.5 grad/grad/s, and for the light maneuverable aircraft this value is still less. This value is determined, in essence, by the frequency characteristics of the system of damper itself and system of control of aircraft (in particular, booster). ENDFOOTNOTE.

The latter in connection with the incidence/drop in the rudder-effectiveness derivative leads to the need for transition/transfer to the completely rotatable all-moving fins on the aircraft, intended to fly with the high supersonic speeds at the high altitudes.

§ 6. Composition of the system of administration and problem of its design.

On aircraft, which fly with small subsonic speeds, problem of control system is the only guarantee of transfer of control signals from pilot (or autopilot) to controls. This problem is implemented by means of the mechanical wiring/run from the control levers in the cabin/compartment to the rudder of aircraft, similar to that shown in Fig. 19.5.

On high-speed aircraft problems of control system substantially are complicated, and its composition in connection with this considerably is expanded. Control systems on these aircraft solves the following problems;

1) transmit control signals from the pilot to the controls (regime of piloting by pilot);

2) transmit control signals from the actuating mechanisms of the systems of automatic control to organ of control;

3) ensure the necessary power for the control displacement;

4) ensure the static and dynamic stability (stabilization);

5) form/shape the necessary handlings;

6) form/shape control signals for automatic trajectory control;

7) form/shape signals to the director instruments during the

regimes of semiautomatic piloting by pilot according to the director instruments;

8) ensure an increase in the safety of piloting with pilot by the corresponding signaling and limiting the control displacement with the approach to the maximum permissible values of the parameters of flight (V, M, n_y, α, γ, β etc.).

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These problems are solved by introduction to system of control of whole series of new mechanical, hydraulic and electrical devices/equipment and automatic systems. Therefore, besides the mechanical wiring/run, the systems of control of contemporary high-speed aircraft include (Fig. 19.20a):

1) the hydraulic amplifiers of power - control actuators (boosters) and hydraulic system of their feed/supply;

2) the system of the load of control levers (during the fully powered controls);

3) the system of the guarantee of the necessary characteristics of static and dynamic stability and controllability (regulators of gear ratios in the wiring/run from the control levers to the rudder, the dampers of oscillation, the automatic machines of stability, the automatic machines of slip, the automatic machines of balancing/trimming, the autothrottles, etc.);

4) the system of director instruments with the calculators of control signals;

5) the system of automatic piloting (autopilot or other automatic

systems, which accomplish the tasks of automatic search and induction/guidance, automatic landing, etc.);

6) the signaling system and limitation of maximum regimes (OPR).

Certainly, on each specific aircraft presence of all enumerated above systems is not compulsory. A question about the need of applying one or the other system is solved in the process of the design of aircraft and its system for control depending on the characteristics of aircraft and its designation/purpose. An example of arrangement/position is shown in Fig. 19.21.

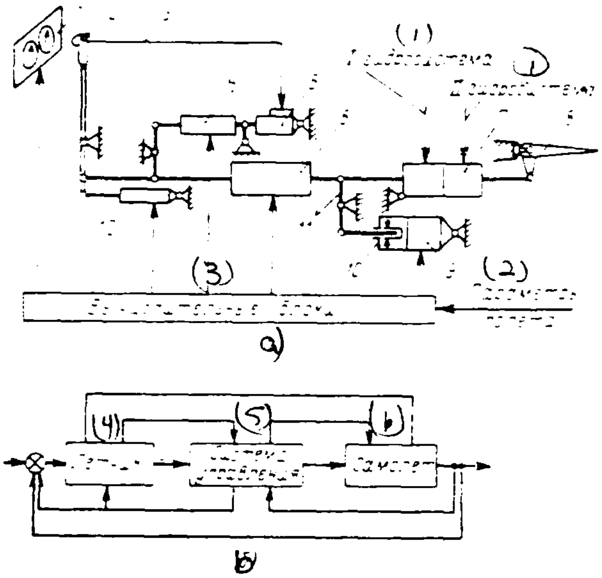

Fig. 19.20.

The exemplary/approximate composition of the system of control of contemporary aircraft (a) and new connections/communications in the control loop (b): 1 - director instruments; 2) control lever; 3) the electrical signal of trimming efforts/forces (balancing/trimming on the efforts/forces); 4 - loader; 5 - mechanism of trimmer effect (MTE); 6 - actuating mechanism of the systems of an improvement in the stability characteristics and control; 7 - two-chamber hydraulic booster with two hydraulic systems of feed of chambers; 8 - control surface (control); 9 - actuating mechanism of the system of trajectory control; 10 - mechanism of turning off mechanism 9; 11 - elements of mechanical wiring/run; 12 - actuating mechanism of the system of the limitation of maximum regimes (OPR). Arrows/pointers showed electrical control signals from the calculating units of automatic systems to the appropriate actuating mechanisms.

Key: (1). Hydraulic system. (2). Flight parameters. (3). Calculating blocks. (4). Pilot. (5). Control system. (6). Aircraft.

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Design of automatic systems of control of aircraft represents fairly complicated independent problem, which in practice of contemporary aircraft construction is solved by specialized organizations for assignments of aircraft-construction OKB. This of problems is the object/subject of the examination of special courses.

In present textbook we will pause only at questions of design of elements of control systems, which participate in process of aircraft handling by pilots. These elements include:

1) mechanical wiring/run;

2) hydraulic boosters from hydraulic system of feed/supply;

3) the system of the load of control levers;

4) the connected to the wiring/run actuating mechanisms of the systems of the guarantee of the necessary stability characteristics and controllability, and also systems of limiting maximum regimes.

During design of part of control in question system it is necessary to approach reaching/achievement of minimum structural weight under condition for fulfilling requirements with respect to guarantee of assigned degree of reliability, technological effectiveness of design and convenience in operation.

§ 7. Initial data for the design of system administrations. Stages of design.

Design of system of control of aircraft begins after determination of its fundamental weight and aerodynamic parameters, and also selection of controls and checking (by calculation or by experimental-purgings) of their efficiency in all possible flight conditions. On obtaining of these data during the first stage of designing the calculations of the required deviations of organs of control and hinge moments on them during different regimes of takeoff, gain of altitude, level flight, maneuvering, reduction/descent, landing approach and landings are performed, including the cases of the failures of engines. The maximum necessary control displacements and the balancing deviations of these organs/controls in different horizontal flight conditions are determined.

In parallel with these problems in electronic analog computers ("models") simulation of processes of motion of projected/designed aircraft during control displacement and external disturbances/perturbations is conducted. The results of this simulation make it possible to determine the necessary means of the automation of control system, to fit the preliminary laws of control, to establish/install the permissible limits of a change in the parameters of the elements of system, with which is ensured the stability of control loop.

Obtained data according to control displacements while different maneuverings, and also on stability conditions for control loop make it possible to determine necessary characteristics of kinematic transfer from levers to controls and characteristic of load of control levers, which ensure accomplishing TTT according to handlings. During calculations of the required characteristics of load are calculated also the requirements on the basis of the limitation of the maximum angles of deflection of controls in some flight conditions, the structural strengths placed by conditions of aircraft.

These limitations are characteristic feature of contemporary aircraft construction, because of which it is possible to substantially reduce weight of structure of aircraft. As an example it is possible to give the most frequently used (especially on the heavy aircraft) limitation of the deflections of rudder at the high speeds (Fig. 19.22), the preventing vertical tail assembly from the destruction under the erroneous effects of pilot and the excessive deflection of rudder.

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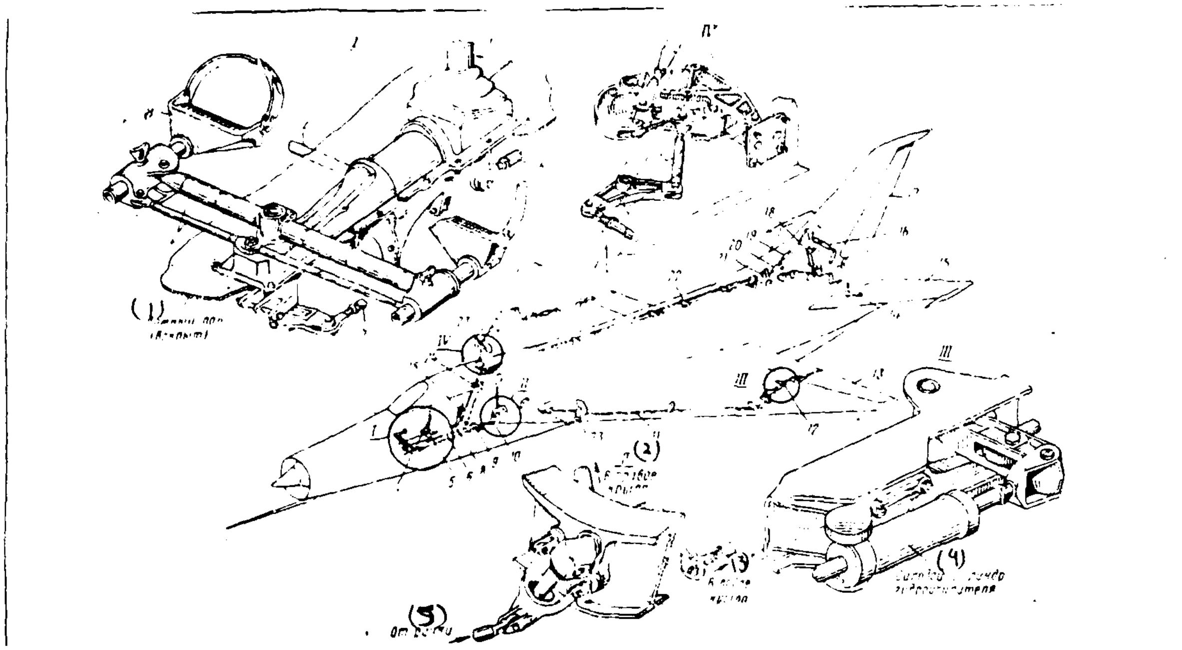

Fig. 19.21. Schematic of arrangement/position of fundamental elements of mechanical feature of irreversible stick-and-power control circuit of controls of light high-speed aircraft (I - central control): 1 - knob/stick of longitudinal and lateral controls; 2 - aileron control rod; 3 - control pedal of rudder; 4 - parallelogram mechanism; 5 - control rod by rudder; 6 - control rod by stabilizer; 7 shielding cover from hitting into wiring/run of foreign objects; 8 - control rod by ailerons; 9 - the loading device of apparitors; 10 - central unit of the distribution of control of ailerons into the right and left wing; 11 - control line by ailerons in the wing; 12 - hydraulic

booster of aileron drive; 13 - aileron (left); 14 - rotational axis of stabilizer; 15 - controllable stabilizer (left half); 16 - hydraulic booster of rudder drive; 17 - rudder; 18 - hydraulic booster of the drive of stabilizer; 19 - mechanism of the trimmer effect of pitch control; 20 - the loading device of pitch control; 21 - mechanism of gear reduction in the pitch control; 22 - roller guard of rod of stabilizer control and by rudder; 23 - mechanism of nonlinear transfer in control of ailerons; 24 - sealed/pressurized outlets of the pull rod from the pressurized cabin; 25 - the loading device of lateral control.

Key: (1). False bottom (it is concealed). (2). In right wing. (3). In left wing. (4). Power cylinder of hydraulic cylinder. (5). From knob/stick.

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Certainly, the strength of tail assembly it would be possible to ensure corresponding strengthening of its construction/design and construction/design of the tail part of the fuselage, but this would require a substantially larger gain in weight, than during the introduction of limitations.

Together with theoretical by studies, conducted, as a rule together by specialists by aerodynamicists, by dynamic loudspeakers, by controllers and by hydraulics specialists, during the first stage of designing control system is produced equipment of flight deck and arrangement/position in it of levers of main control, levers and toggle switches of synchro control, control panels, signaling and control/check of operation of hydraulic system and means of automation. These works, just as the separator of the routes of wirings/runs of main and synchro controls, arrangement and connecting of the fundamental assemblies of the system of control and elements of the construction of the aircraft, are carried out by the designers of different brigades in essence in the full-scale mock-up of future aircraft.

All enumerated above studies compose basis of preliminary design of system of control, entering as component part preliminary design of new aircraft. The preliminary design, which completes the first stage of the design of control system, it contains:

1) the short description of designation/purpose and the composition of system, its elements and their interaction, controls, levers and other elements of control and signaling in the cabin/compartment, the description of the principles of an increase in reliability and providing safety;

2) initial data for design of the system (balancing control displacements in different flight conditions, including failures, required deviations and speeds the cross bars of controls while maneuverings and stabilization, a change in the hinge moments on the controls during different flight conditions, the inertial characteristics of controls, the required damping characteristics of controls for the guarantee necessary of flutter characteristics);

3) the schematic diagrams of control (along all channels of main and synchro controls) with the included in them actuating mechanisms

of the systems of load, automation and the hydraulic boosters;

4) the kinematic dependences of the control displacements from the deviation of control levers;

5) the characteristic of the load of control levers (including limitations);

6) performance calculations and the selection of the hydraulic boosters of the drive of control surfaces (serial or new, ordered specially for this aircraft);

7) the principal diagram and the calculation of power of the hydraulic system of feed/supply of hydraulic boosters, including different cases of failures;

8) schematic diagrams and the laws of the work of automatic systems;

9) connecting the arrangement of the fundamental assemblies of control system with the elements of construction/design and other systems of aircraft;

10) the layout of levers and other elements of control system in the cabin/compartment.

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Preliminary design taking into account observations of simulated board becomes the basis itself of design development of elements of control system.

Second development stage of design of control system includes works let us rub to directions:

1) structural/design development and issue of working drawings to elements of control system, developed/processed in aircraft-construction OKB (as a rule this mechanical wiring/run with included in it different mechanisms, load system and installations of hydraulic system);

2) theoretical and structural/design study in specialized organizations, ordered assemblies and systems, manufacture of test samples and their testing;

3) development and construction of bench installations for complex final adjustment of control system.

Latter/last direction as a result of extreme complexity of systems of control and need for experimental final adjustment of many constructive solutions includes development of whole complex of bench installations from full-scale stand, which includes all actual elements of control system and which makes it possible to master questions of their interaction, to small bench installations, intended for solution of separate structural/design problems.

Special position among works of this occupy creation of stand-trainer, intended for final solution of questions of selection of optimum handlings, training aircrew for first flight, final adjustments of methodology of piloting in different flight conditions, including failures, etc.

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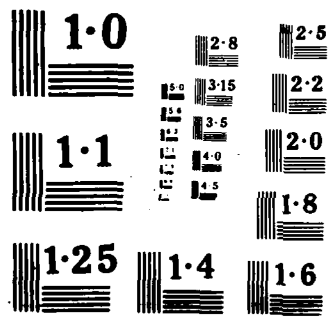
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Finally, third stage includes in essence experimental development of systems and complexes on stands, and then on constructed aircraft, and implementation of corresponding modifications. Among these works important place occupy works on plotting of frequency curves/characteristics of systems and separate assemblies, testing the stability of ducts/contours and elimination of possible auto-oscillations, research of the work of control system in the case of different failures.

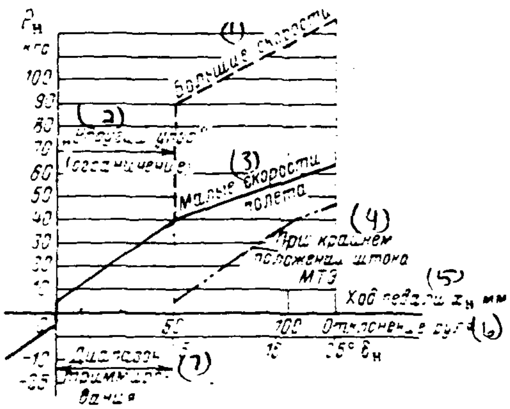

Fig. 19.22. Characteristic of load of control pedals of rudder of heavy aircraft with limitation of angles of deflection of control at high speeds (in wiring/run it is used nonlinear transfer from pedals to control).

Key: (1). High speeds. (2). "Elastic support" (limitation). (3). Low flight speed. (4). In extreme position of rod. (5). Course of pedal. (6). Rudder deviation. (7). Range/trimming.

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§ 8. Arrangement of levers and other elements of control in the cockpits.

Control of aircraft during piloting by pilot is accomplished/realized with the aid of displacement/movement of corresponding control levers. In the light practice it was established that the longitudinal and lateral controls on the light maneuverable aircraft are accomplished/realized with the aid of the respectively longitudinal and cross travels of the control stick, established/installed between the elbows of pilot (Fig. 19.23, and also 19.5, 19.9 and 19.21), and on the heavy and nonmaneuverable aircraft - with the aid of control stick , established/installed on the column, which sways in the longitudinal vertical plane of the aircraft (see Fig. 19.26a, 19.27 and 19.31). Sometimes handwheel is established/installed on the shaft, passing through the instrument panel and which has the capability of progressive movements in the guides along the longitudinal axis of aircraft (Fig. 19.24). Lateral control on the nonmaneuverable aircraft is accomplished/realized by rotations of handwheel.

These control levers are utilized also with changes in direction of flight - turns.

Countering or compiling of slip angle with turns, cross wind or unsymmetric rod/thrust on aircraft of all types is accomplished/realized by bias/displacement of pedals.

For facilitating control process kinematics of guide is always projected/designed in such a way that direction of motion of control levers would coincide with necessary direction of rotation of aircraft.

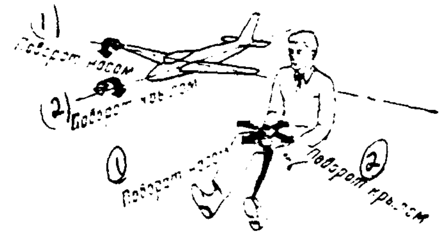

Fig. 19.23. Installation of control stick of light maneuverable aircraft along pitch (nose rotation) and bank (rotation by wing). Key: (1). Rotation by nose. (2). Rotation by wing.

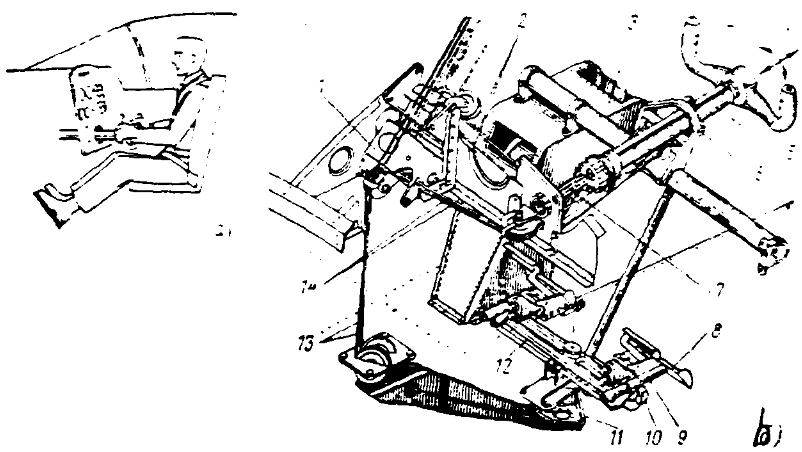

Fig. 19.24. Installation of handwheel on shaft, passing through instrument panel of pilot: a) general view of installation; b) construction/design of installation of light aircraft with cable control: 1 - bracket with rollers, 2 - control cables of ailerons; 3 - central control panel (throttle controls are not conditionally shown); 4 - handwheel; 5 - tubular shaft of handwheel; 6 - rollers; 7 - shaft guide during pitch control (it is established/installed in bearings and simultaneously it serves for transmission of rotation of handwheel during roll control); 8 - control pedal of rudder; 9 -

opening/aperture for control of installation of pedal; 10 - index; 11 - sector; 12 - parallelogram mechanism; 13 - control cables of rudder; 14 - control cables of elevator.

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Thus, for example, during shifting of stick (or the column of handwheel) forward "from itself" the nose of aircraft begins to lower down, overloading falls, speed grows. During shifting of stick "to itself" the nose of aircraft begins to be built up upward, overloading grows, speed falls. With the deflection of the handle (rotation of handwheel) to the right the aircraft also begins to roll to the right, and the forward movement of right pedal causes the motion of the nose of aircraft also to the right and so forth.

Fig. 19.25 depicts rules of signs accepted for deviating of controls, control levers and force feel.

On aircraft, for which during flights there will be characteristically prolonged action of sufficiently heavy overloads, control with the aid of usual control levers can prove to be inconvenient. Therefore for control on such aircraft (for example, hypersonic or VKS) can be utilized some other control levers, for example, the side stick, which makes it possible for the hand of pilot to lie/rest on the elbow-rest (see Fig. 19.12).

So that control would be convenient and would not cause fatigue of pilot, control control levers must be arranged/located on specific distances from seat, instrument panel and from each other. Taking into account different growth of pilots, the seats of pilots are fulfilled by those adjusted. By those adjusted are also made pedals.

Fixed dimensions of levers of main control are manufactured by practice and are fastened by appropriate recommendations. An example of such recommendations for the main-line passenger aircraft is shown in Fig. 19.28a [1], and for military aircraft in Fig. 19.28b [5], page 208.

FOOTNOTE [1]. In the present paragraph are utilized the materials of the experiments of V. K. Nefedov, published in the articles the "flight deck of aircraft and comfort" and "aesthetics and the controls of aircraft" (see journal "Technical aesthetics", No 11, 1966 and No 10, 1968). ENDFOOTNOTE.

Distance between pedals (on their centers) for aircraft, whose longitudinal and lateral controls are accomplished/realized by knob/stick, is accepted order 420-460 mm so that legs of pilot would not prevent deflections of the handle. On the aircraft with the steering-wheel control this distance it is possible to accept somewhat less (order 320-350 mm). The width of pedal itself is taken as the equal to 150 mm.

Post of pedal operation on light aircraft, as a rule is fulfilled with vertical rotational axis of parallelogram mechanism, on which pedals (see, for example, Fig. 19.21 and 19.24b) are established/installed.

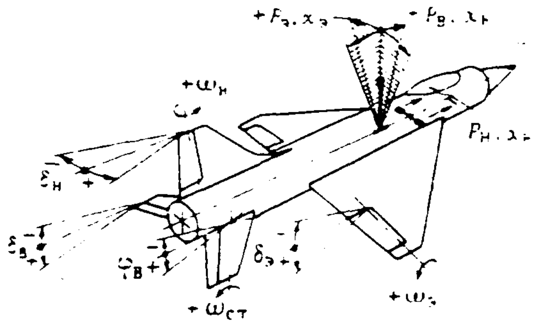

Fig. 19.25. Rules of signs for deviating of controls, control levers and force feel x - linear displacement of control lever:

$P_{э} = P_{н\,пр} - P_{н\,лев}$, $P_{н} > 0$, if $P_{н\,пр} > P_{н\,лев}$.

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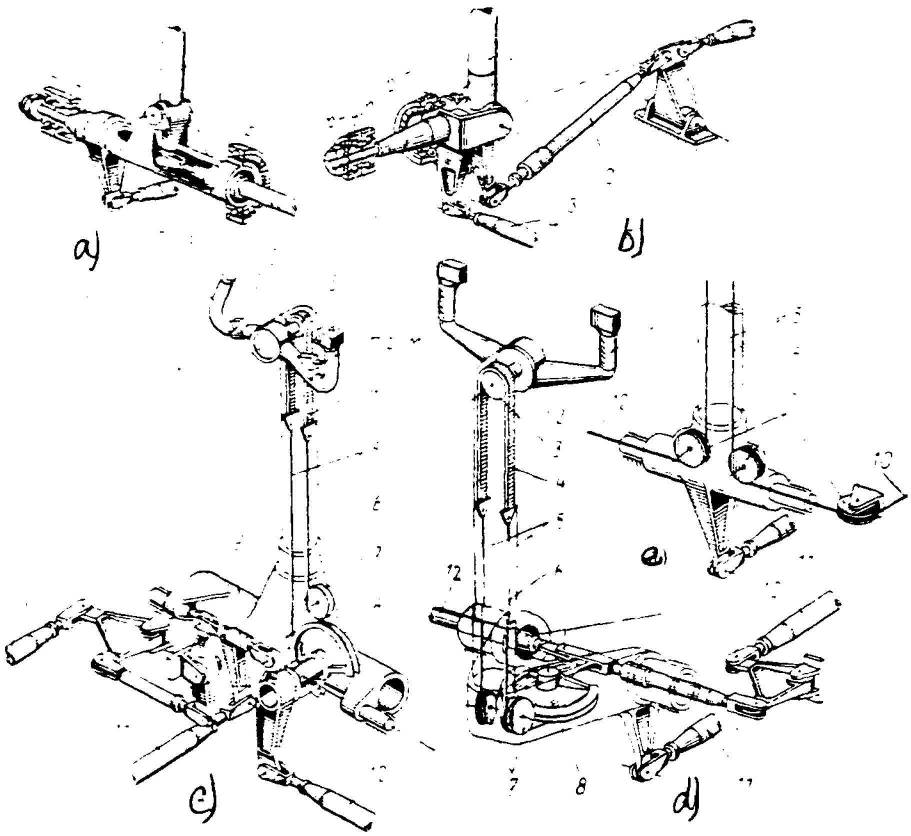

Fig. 19.26. Fundamental methods of guaranteeing independence of longitudinal and lateral controls during planning of central controls: a and b) control of knob/stick (light maneuverable aircraft): 1) control stick 2 - control rod on the pitch; 3 - repair link, which ensures the possibility of the turning of the rod tip; 4 - control rod along the bank; 5 - axis-bracket of the installation of knob/stick (r.p. - radial bearing, RUP - radial-thrust bearing); c, d and e) the diverse variants of the outlet of the pull rod or control cable along the bank during control of the handwheel: 1 - handwheel; 2 - rotational axis of handwheel; 3 - gear ("chain wheel"); 4 - Gall's

circuit; 5 - control cables along the bank; 6 - column; 7 - roller; 8 - sector; 9 - rod/thrust with the hinge; 10 - rod/thrust (or cable) of roll control; 11 - control rod on the pitch; 12 - connection/communication with the second handwheel (tr.m. - tumbler of control of trimmer mechanism in the channel of pitch).

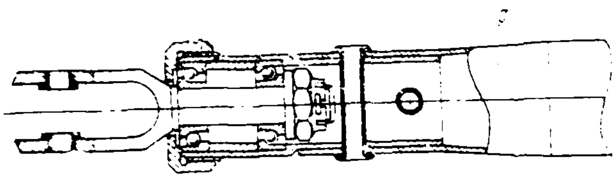

Fig. 19.27. Constructions/designs of hinged element of rod/thrust: 1 - being pulled forked cap; 2 - radial thrust ball bearings; 3 - tube of thrust.

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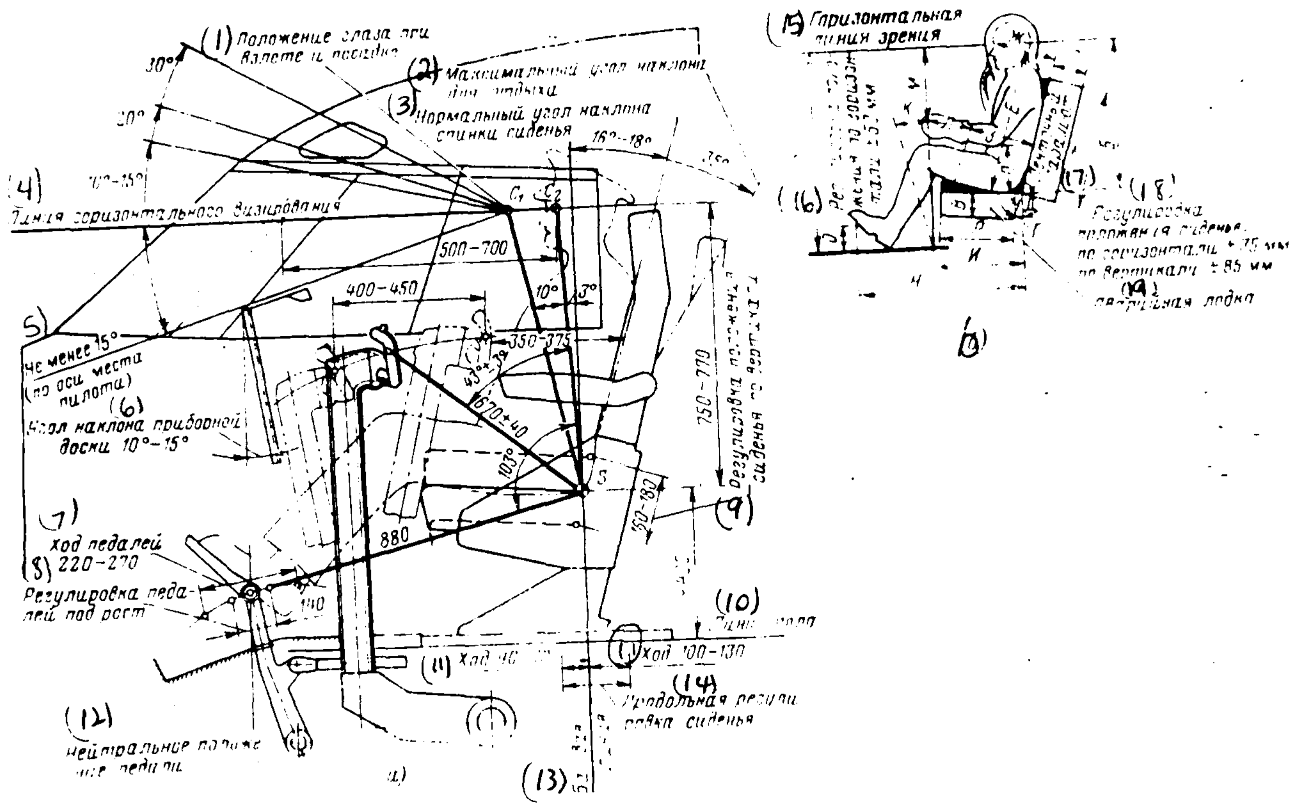

Fig. 19.28. Recommendations regarding arrangement/position of levers of main control in cockpits of main-line passenger aircraft (a) and aircraft ejection seats (b).

Key: (1). Position of eye during the takeoff and the fitting/landing. (2). Maximum slope angle for rest. (3). Normal angle of slope of backrest. (4). Line of horizontal sighting. (5). Not less than 15° (along axis/axle of place of pilot). (6). Angle of slope of instrument panel of 10°-15°. (7). Course of pedals 220-270. (8). Control of pedals under increase/growth. (9). Control of position of seat on vertical line. (10). Line of floor. (11).

Course. (12). Neutral position of pedal. (13). Base line. (14). Longitudinal control of seat. (15). Horizontal line of view. (16). Control of position on horizontal ±50 mm. (17). Ribbon parachute. (18). Control of position of seat: on horizontal ±35 mm; on vertical line ±85 mm. (19). Emergency boat.

А	Б	В	Г	Д	Е	Ж	З	З*	И	К	Л	М	М*	Н	О	П	Р	С	Т
925	810	125	[illegible]	101	740	250	300	430	475	150	225	200	[illegible]	900	[illegible]	230	[illegible]	[illegible]	[illegible]
975	822	125	[illegible]	101	760	244	[illegible]	[illegible]	[illegible]	150	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
1025	840	125	[illegible]	101	[illegible]	244	[illegible]	[illegible]	[illegible]	150	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
1075	846	125	[illegible]	101	[illegible]	250	[illegible]	[illegible]	[illegible]	150	[illegible]	[illegible]	[illegible]	[illegible]	125	[illegible]	[illegible]	[illegible]	[illegible]

Z* and M* - are given for control of handwheel. Remaining sizes/dimensions for the administrations of knob/stick and handwheel are identical (sizes/dimensions they are given in mm).

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On the heavy aircraft the pedals are fulfilled by suspension or lever/crank with the horizontal rotational axes (see Fig. 19.31).

During planning of posts of manual control it is necessary to ensure independence of longitudinal and lateral controls. The structural/design realizations of this requirement there can be set; however, the most widely used methods during the installation of knob/stick they are:

1) the outlet of the pull rod of control according to one of the channels (pitch or bank) along the theoretical rotational axis of knob/stick during control along another channel (Fig. 19.26a);

2) the outlet of the pull rod of control according to one of the

channels on forming the cone with the axis/axle, which coincides with the rotational axis of knob/stick during control along another channel; the rotational axis of knob/stick and thrust axis with this method must intersect at the point (apex of the cone), structurally combined with the attachment point of rod/thrust to the following rocker (Fig. 26b, by this method it is provided the independence of administrations on the diagrams in Fig. 19.5 and 19.21).

If on aircraft steering-wheel control is established, then, as a rule is utilized first method. If during control of the knob/stick of different constructions/designs through the appropriate axis/axle both control rod along the bank and control rod on the pitch could be derived/concluded, then during the steering-wheel control this conclusion is accomplished/realized only by a rod/thrust or a control cable along the bank through the rotational axis of column (pitch control) (see Fig. 19.26c, d, e and Fig. 19.31).

It must be noted that with all methods of guaranteeing independence of administrations, rod/thrust, relative to axis/axle of which occurs rotation of control lever, to avoid its torsion must have structural element, which allows/assumes turning of one of caps relative to another (Fig. 19.27).

Deviations of control levers (just as rockers in guide) it is accepted to measure on linear displacements (chords) of their calculation points. The course of pedals usually is accepted within the limits ±90-135 of mm, the course of the lever of pitch control 280-480 mm, the knob stroke during control by ailerons 100-150 mm (here larger values of courses, as a rule they correspond to heavier aircraft). The maximum angle of the turn of handwheel (with a calculated radius of 250-300 mm) on the nonmaneuverable aircraft at present is made in limits of ±70-±90°, while in the old heavy machines from boosterless control for guaranteeing the acceptable efforts/forces on the handwheel it was necessary to obtain satisfaction this angle to ±180°. This led to the inconvenient cross position of the hands of pilot if necessary for the large aileron deflection.

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High value has also selection of correct form and sizes/dimensions of handwheel, since unsuccessful form of handwheel can lead to difficulties in piloting as a result of interfering for elbow (Fig. 19.29).

Maximum deviations of control levers must be limited compulsorily to those assembled in control stations of adjustable travel limiters - stops.

It is necessary to remember that when, on aircraft, two pilots are present, their control stations kinematically are united so that control levers of left and right pilots would diverge synchronously (see, for example, Fig. 19.31).

During planning of control in cabin/compartment and, in particular, control stations, besides fulfilling of requirements on strength, rigidity and minimum weight, should be focused attention also to technological effectiveness of design. As the practice of planning shows, control in the cabin/compartment with loaders and numerous mechanisms (see, for example, Fig. 19.21 and 31) is obtained by one of the most complicated units of the mechanical feature of control system, the installation, control and plotting of curves/characteristics of which under the squeezed conditions of the pilot's cabin presents known difficulties.

But if questions of technology are thought over at early stages of planning, then aggregates/units of control in cabin/compartment can be arranged in such a way that their assembly, installation and plotting of curves/characteristics can be carried out under open and convenient conditions out of pilot's cabin, and then entire unit in collection can be will be established/installed to aircraft with the aid of sufficiently small number of fasteners.

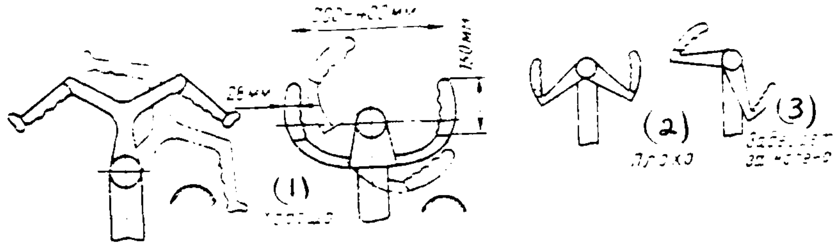

Fig. 19.29. To selection of form of handwheel.

Key: (1). It is good. (2). It is bad/poor. (3). It interferes for elbow.

(a) № по пор.	(b) Органы управления	(c) Координаты органов управления					
		(d) левое сиденье			(e) правое сиденье		
		x	y	z	x	y	z
1	Штурвал управления рулем высоты и элеронами (нейтраль)	505	[illegible]		50[illegible]	[illegible]	
2	Педали управления рулем направления (нейтраль)	8[illegible]	9[illegible]	1[illegible]	[illegible]	9[illegible]	[illegible]
3	Рычаг управления двигателем (РУД)	39[illegible]	[illegible]	[illegible]	39	5[illegible]	[illegible]
4	Рычаг аварийного торможения колес	[illegible]30	[illegible]	[illegible]	73	[illegible]	[illegible]
5	Кнопка аварийного отключения автопилота	(5a) Правая рукоятка штурвала			(5b) Левая рукоятка штурвала		
6	Рычаг выпуска и уборки шасси	530	−90	[illegible]	53[illegible]	−90	33[illegible]
7	Рычаг выпуска, уборки закрылков и предкрылков	540	−140	500	54[illegible]	−140	[illegible]
8	Рычаг аварийного выпуска шасси	—	—	—	[illegible]	640	−42[illegible]
9	Орган управления передним колесом	(9a) Педали ножного пульта			(9a) Педали ножного пульта		
10	Рычаг управления интерцепторами	395	660	410	400	6[illegible]0	590
11	Щиток управления автопилотом	345	730	355	345	730	64[illegible]
12	Тумблер триммера загружателя руля высоты	(5b) Левая рукоятка штурвала			(5a) Правая рукоятка штурвала		
13	Рычаг триммера загружателя руля направления	650	140	400	650	140	[illegible]
14	Рычаг триммера загружателя элеронов	650	14[illegible]	550	650	140	4[illegible]

Fig. 19.30. Recommendations regarding arrangement/position of levers and other elements of control in cockpit of main-line passenger aircraft taking into account convenience in attainability.

Key: (a). No on pores. (b). Controls. (c). Coordinates of controls. (d). left seat. (e). right seat. (1). Handwheel of elevator control and of ailerons (neutral). (2). Control pedals of rudder (neutral). (3). Engine-control lever (RUD). (4). Lever of emergency braking of wheels. (5). Knob/button of emergency cutoff/disconnection of autopilot. (5a). right lever of handwheel. (5b). the left lever of handwheel. (6). Exhaust gear rod and landing gear retracting. (7). Lever of issue, retraction of flaps and slats. (8). Lever of emergency release of chassis/landing gear. (9). Control of front/nose wheel. (9a). Pedals of foot panel. (10). Control lever of interceptors/spoilers. (11). Control panel of autopilot. (12). Toggle switch of trim tab of loader of elevator. (13). Lever of trim tab of loader of rudder. (14). Lever of trim tab of loader of ailerons.

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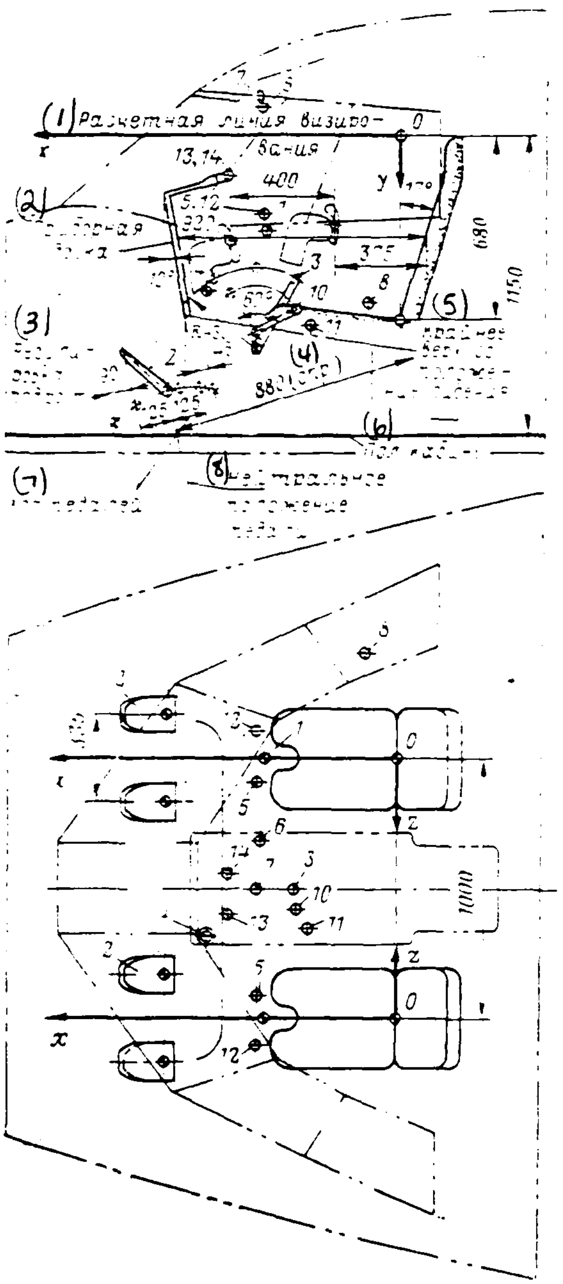

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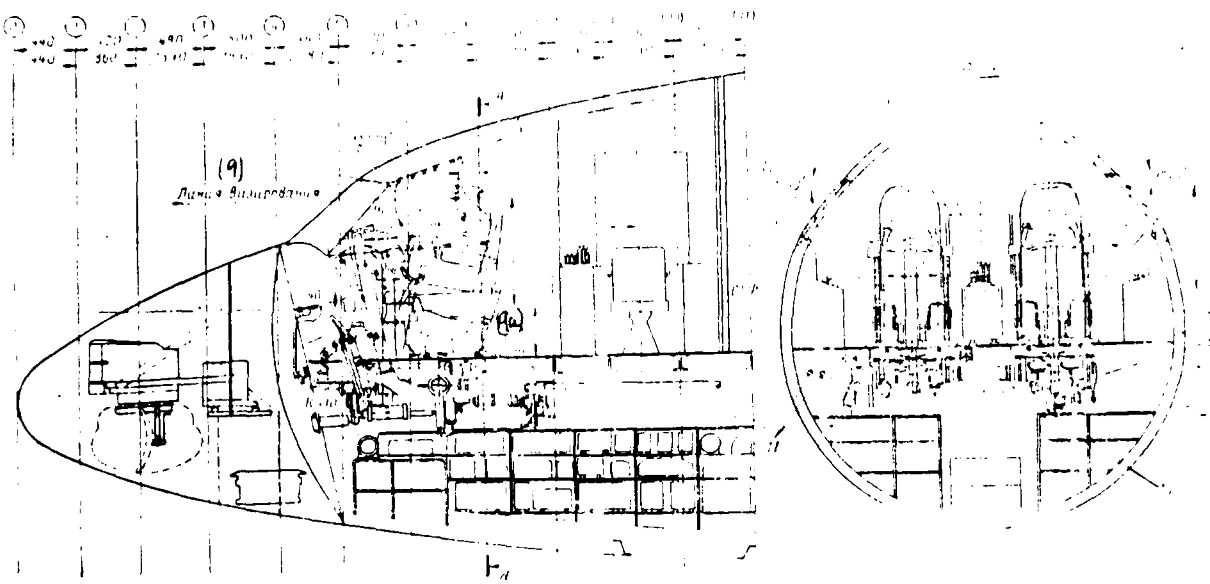

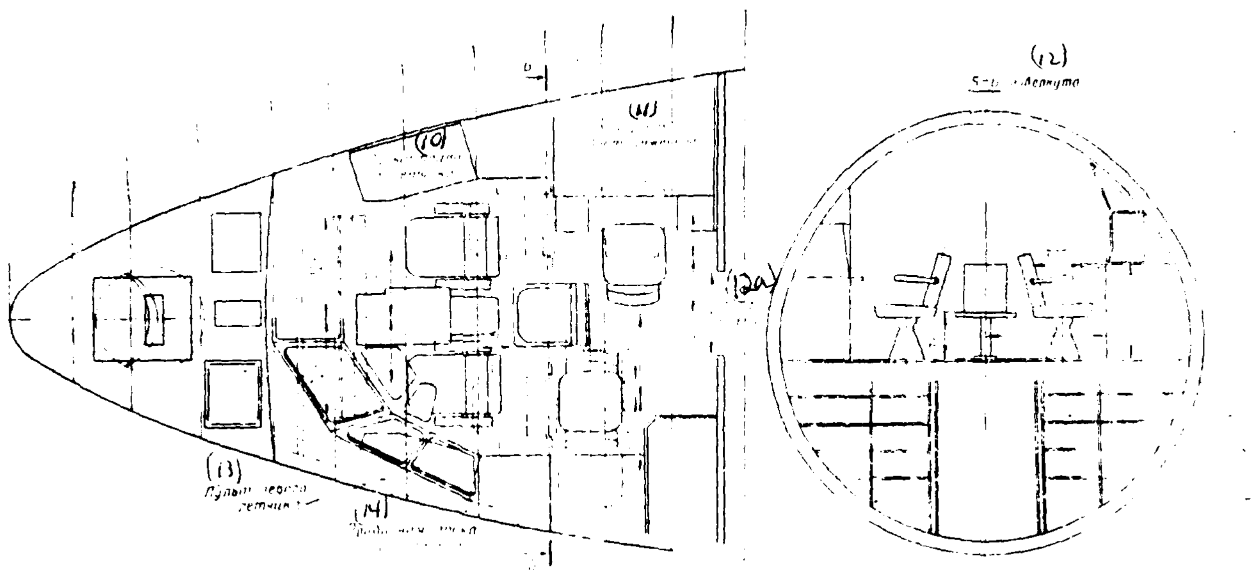

Fig. 19.31. Layout of forward fuselage and flight deck of contemporary main-line passenger aircraft: 1 - left panel for pilot; 2 - pedal of left sensor (craft commander); 3 - handwheel of left-seat pilot; 4 - instrument panel of pilots; 5 - central control panel for pilots with RUD; 6 - handwheel of co-pilot; 7 - pedal of co-pilot; 8 - left panel; 9 - elements of control line by floor of cabin/compartment; 10 - thrust of connection of control levers of right and left of pilots; 11 - output of rods/thrusts to controls; 12 - panel of flight engineer with RUD.

Key: (1). The reference line of vision. (2). Instrument panel. (3). Control young forest. (4). (spr.). (5). Extreme upper position of seat. (6). Floor of cabin/compartment. (7). Course of pedals. (8). Neutral position of pedal. (9). Line of sighting. (9a). Floor of cabin/compartment. (10). Panel for co-pilot. (11). Panel edge. of engineer. (12). it is turned. (12a). Axis/axle of aircraft. (13). Panel for left-seat pilot. (14). Instrument panel of navigator.

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With good technological effectiveness of design it is possible to go even to a certain gain in weight, which will be paid by improvement in the quality with the assembly, by ease/lightness of operation, and consequently, by an increase in the reliability.

Fig. 19.30 as example gives some recommendations regarding arrangement/position of levers and other elements of control in cockpit of contemporary main-line passenger aircraft, while in Fig. 19.31 - layout of nose section of this aircraft. As can be seen from Fig. 19.31, the elements of control system in the a~ea of cockpit occupy the sufficiently large space, which must be provided for with the layout of contemporary aircraft.

§ 9. Design of mechanical guide.

On aircraft with boosterless control mechanical guide connects control control levers of aircraft with its organs of control - by control surfaces. On the aircraft with the assisted control mechanical guide conditionally can be divided into two parts:

1) non-load-bearing - from the control control levers to the intake rockers of hydraulic boosters (boosters), that receives the efforts/forces only of the hands (or legs) of pilot or loader;

2) power - from the output component/link of booster to the control surface, that receives and transmitting to the hydraulic booster of load from the hinge moment on the control surface.

At present guide they attempt to project/design so as to virtually completely exclude its powering unit, and output component/link of booster to connect directly to lever of control surface.

To mechanical guide are connected up devices/equipment of load of control levers (in irreversible servo-controlled system), actuating mechanisms of systems of improvement in stability characteristics and controllability and automatic control (autopilot).

Of dependence on layout conditions, weight, rigid and some other factors mechanical guide on contemporary aircraft can be projected/designed three fundamental forms:

1) with rigid, with which control signals are transmitted to control surfaces with the aid of reciprocating displacements/movements of tubes, working in compression and extension (Fig. 19.32a);

2) by bending, during which control signals are transmitted with the aid of reciprocating displacements/movements of cables or steel tapes, working only in extension, in connection with which flexible guide compulsorily it must consist of two branches (straight line and

recurrent, Fig. 19.32b);

3) rotary, presenting variety rigid guide, in which signals are transmitted by reversible rotary motions of tubes - shafts, however deviation of control surfaces are accomplished/realized with the aid of spiral ball bearing converters of rotary motion into progressive/forward (Fig. 19.32c).

Frequently are used combinations of different types of guides. As a rule this of the combination of the rigid of progressive/forward and cable or rigid progressive/forward and rotary guides.

Basic purpose of guide - with smallest distortions (in phase and amplitude) to transmit control signals from pilot or automatic system to control surface.

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General/common recommendation regarding design of elements of guide can be formulated as follows:

- it is necessary to attempt to replace elements, which work by bending and torsion, with elements, which receive only longitudinal forces (Fig. 19.33). Especially one should avoid translation of motion with the aid of the rockers (or sectors), connected by the long shafts, which work for the torsion (Fig. 19.33b). Furthermore, it is necessary to focus special attention on the guarantee of rigidity of fastening guide to the construction of the aircraft, since from the rigidity of this fastening to a considerable degree the total hardness of guide depends.

Friction in guide worsens/impairs its frequency characteristics and handlings. With the large friction in the guide, which calls the need for application/appendix to the control levers of sufficiently large forces for their moving, the control of aircraft generally can become impossible. Therefore in the general technical requirements (OTT) for the aircraft depending on their type are specified the maximum permissible frictional forces in the guide, led to the control levers.

Presence of combined types of guide on contemporary aircraft is explained by tendency of designer-design engineers to maximally use the advantage, also, as far as possible to get rid of shortcomings in guides of different forms.

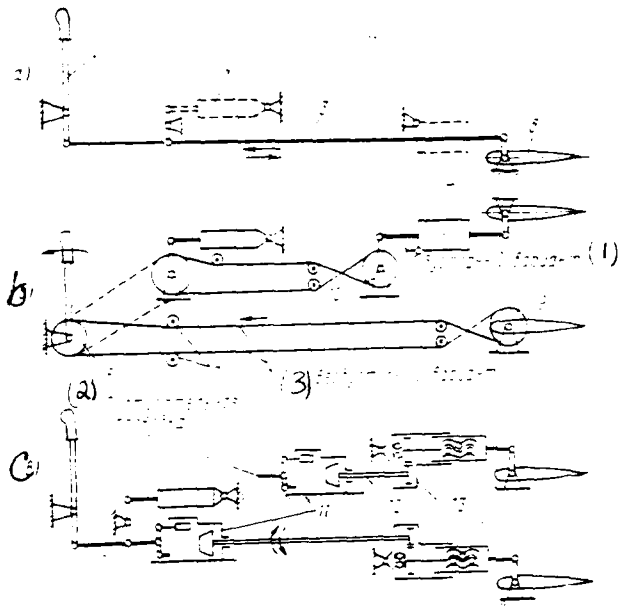

Fig. 19.32. Different types of mechanical guide: a) rigid with recurrent-recurrent-progressive/forward motion of rods/thrusts (by dotted line it is shown version with inclusion in system of necessary hydraulic booster of progressive/forward action): 1 - control lever; 2 - system is load; 3 - control rod; 4 - irreversible hydraulic booster; 5 - control surface; b) flexible (is above shown the version of the use of flexible guide in the irreversible stick-and-power control circuit): 6 - leading sector; 7 - rollers; 8 - cables (or tapes); 9 - driven/known sector; 10 - sector with the lever (by arrow/pointer it is shown the straight/direct branch of guide, which transmits control signal, during the prescribed/assigned motion of control lever): c) the diverse variants of the rotary guide: below - hydro-engine drive is established/installed in the forward fuselage (in cockpit), guide along the fuselage and the wing - rotary

(transmission); above - hydro-engine drive is established/installed directly on the control surface, guide along the fuselage to wing - progressive/forward; 11 - hydraulic booster of rotary action - hydro-engine drive; 12 - rotary guide (transmission); 13 - ball bearing converters of motion (hoists).

Key: (1). Booster version. (2). Progressive/forward guide. (3). Boosterless version.

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Thus, for example, the indisputable advantages of rigid guide with forward motion of rods/thrusts in comparison with the cable run is its simplicity in the operation and the smaller friction with repeated changes in the direction of motion on the rockers. At the same time cable run in the straight/direct sections is obtained it more easily and occupies less than the place, facilitating its layout; cable run more simply it is duplicated/backed up, which is especially important for the aircraft of military designation/purpose. Therefore for the contemporary aircraft the characteristically sufficiently frequent application of rigid guide in the places, where on the conditions of guaranteeing the kinematic constraints between the elements of the system of control and their layout on the aircraft numerous connections are necessary and changes in the direction of the route of guide (for example, in the cockpits), and long straight/direct sections without a change in the direction frequently are made with the aid of the cable run.

However, if designs allow/assume this, then application of rigid guide seems more preferable, since cable run with identical of quantity of transition elements possesses high friction and requires constant attention in operation.

During design of rigid guide one should attempt to maximally reduce number of coupling elements, since each connection increases gap in guide and weight of coupling and holding elements. However, at the excessively large length of their rods/thrusts it is necessary to make with very thick from the condition of retaining/maintaining the stability during the compression [1].

FOOTNOTE [1]. Critical force of compression for the rod/thrust can be determined according to the known Euler formula $P_{крит}=\frac{\pi^2 EI}{\mu l^2}$ (where for the rods/thrusts μ=1). In the practice the calculated and experimental curves of breaking stresses and efforts/forces for the cruxes of the tubular section/cut of different length more frequently are utilized. See, for example, [1], page 286-307. ENDFOOTNOTE.

Therefore the weight of rods/thrusts will rapidly grow/rise and, although in this case the weight of the coupling and holding elements decreases, the total weight of guide will also grow/rise. It is obvious that for each guide depending on the transmitted efforts/forces there is an optimum length of rods/thrusts in the straight/direct sections, at which the weight of guide is smallest (usually this the length of order 1200-1500 mm).

In rigid guide as holding elements guide rockers or roller slide (Fig. 19.34) are used. During the application of guide rockers friction in the system is obtained smaller, but route requires somewhat more than place, since rods/thrusts during the transmission of control signals are displaced not only along their axis/axle, but also in parallel to it.

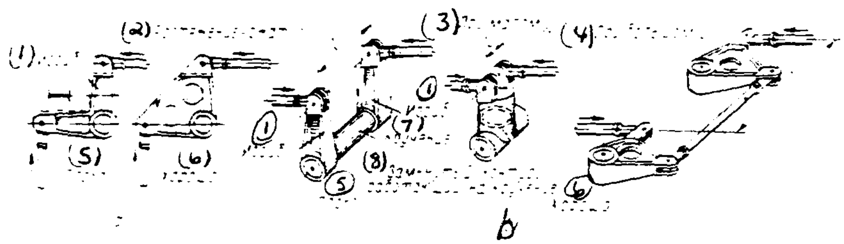

Fig. 19.33. Some recommendations regarding increase in rigidity of elements of guide during design: a) planar rockers; b) three-dimensional/space rockers.

Key: (1). Bending. (2). extension-compression. (3). With low l, (4). With large l. (5). It is bad/poor. (6). It is good. (7). Torsion. (8). Closed contour/outline, which works for torsion.

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For warning/preventing of undesirable differentiality and essential gear reduction in guide with deviations from neutral position during design of guide it is necessary to approach that so that in neutral position angles between axes/axles by lever of rockers and connected up to them rods/thrusts would be equal to 90°, and values of lever arms were selected in such a way that angles of maximum deviations of these levers on would exceed ±30-35°. Only in the control line by elevator or by the controllable stabilizer, which has different course into one and another of side as a result of the large difference of the required elevator angles upward and downward, the axes/axles of the levers of rockers in the neutral position "are shot down" from the normal to the thrust axis of the side of the smaller course, which corresponds to the down-elevator deflection (Fig. 19.35).

During design of mechanical guide of contemporary aircraft, especially heavy with long fuselages, it is necessary to provide for structural/design measures for eliminating effect of linear deformations of fuselage and wing. These deformations are caused in essence by three reasons:

1) by the effective on the construction/design external loads;

2) by a change in the ambient temperature (but this change for the contemporary supersonic aircraft it can occur in limits of -60-+250°C);

3) by the action of overpressure in the pressurized cabin of passenger aircraft.

As show calculations and practical measurements, change in length of fuselage under action of these factors can reach very high values (so, for example, for aircraft of type "Concorde" - order 120 mm).

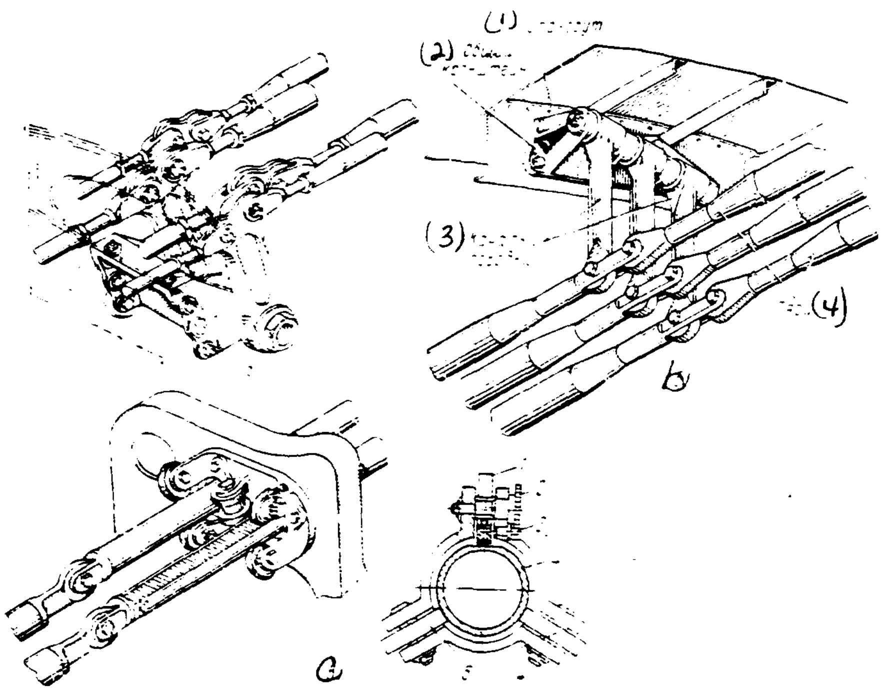

Fig. 19.34. Supporting elements rigid guides with forward motion of rods/thrusts: a) connection of rods/thrusts on dual lugs of holding rockers (carriers); b) connection of rods/thrusts on dual caps, carriers - with identical lugs; c) roller slide; section/cut one rod/thrust and section/cut of upper roller, which ensures control of angle is to the right shown; 1 - roller; 2 - eccentric cylinder-regulator; 3 - catch; 4 - rod/thrust; 5 - lower fixed rollers.

Key: (1). Frame/former. (2). General/common bracket. (3). Rockers (carriers). (4). Rods/thrusts.

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Therefore, if we do not accept special structural/design measures for the compensation for the linear structural distortions of aircraft, then this can lead to the limitation of the possible deflections of control. This will occur because almost entire course of rigid control line will be already selected only for retaining/maintaining the neutral surface position, and control levers will be in this case close to the end positions, limited by stops, or even they will become to these stops. Thus, for example, on one of the passenger aircraft before the introduction of special compensating mechanism (Fig. 19.36b) into the control line by rudder the bias/displacement of pedals for retaining/maintaining the neutral surface position during the deformations of fuselage in flight comprised order 100 mm.

If cable run is used, then due to presence of its two branches of structural distortion of aircraft they do not lead to "shears" in control (i.e. to disagreement/mismatch of surface position and control levers), but cables in this case can either be weakened/attenuated and take up the sag or strongly be tightened. In the first case in control appears large gap, the secondly - strongly increases the friction. In both cases control can become impossible.

For compensation for linear structural distortions of aircraft in rigid guide are used compensating rockers and mechanisms (see Fig. 19.36a, and b), and in cable - control mechanisms of tension of cables (see Fig. 19.36c). One should note that during the existing constructions/designs the mechanisms of the tension of cables have diameter on the order of 300-400 mm, that it is necessary to consider with the layout of aircraft, leaving for them the appropriate spaces.

If rotary guide is used, questions of compensation for structural distortions of aircraft are easily solved by application of slot telescopic shafts.

In conclusion it is necessary to note that structural/design difficulties of guaranteeing necessary characteristics of mechanical guide, complexity of its layout and large weight on contemporary aircraft, especially heavy, increasingly more frequently make it necessary to turn to study of problems of transition/transfer to long distance electric control. This system is at present used, in particular, on the French aircraft "Mirage" III and "Mirage" IV for control of rudders, and is also the primary (duplicated/backed up for the increase safety) system for control on English-French SPS "Concorde".

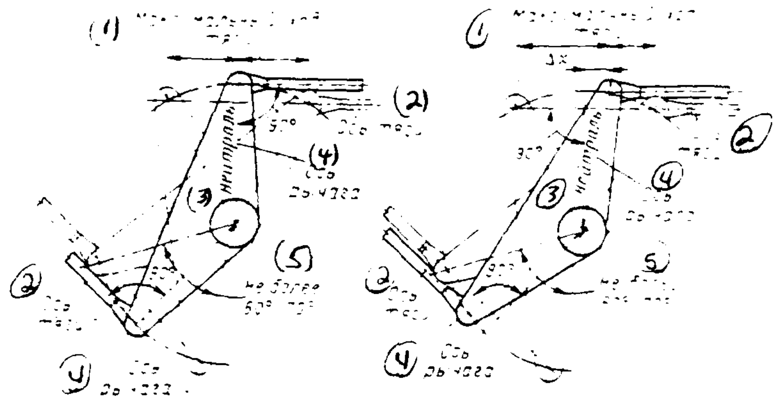

Fig. 19.35. Installation of rods/thrusts with respect to axes/axles of levers of rockers. Is to the right shown "beating" of neutral (Δx) of levers in the control line by elevator (controlled by stabilizer) to the side of smaller course.

Key: (1). Maximum course of rod/thrust. (2). Thrust axis. (3). Neutral. (4). Axis/axle of lever. (5). Not more.

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Mechanical guide available on this aircraft is the third (emergency) system for control.

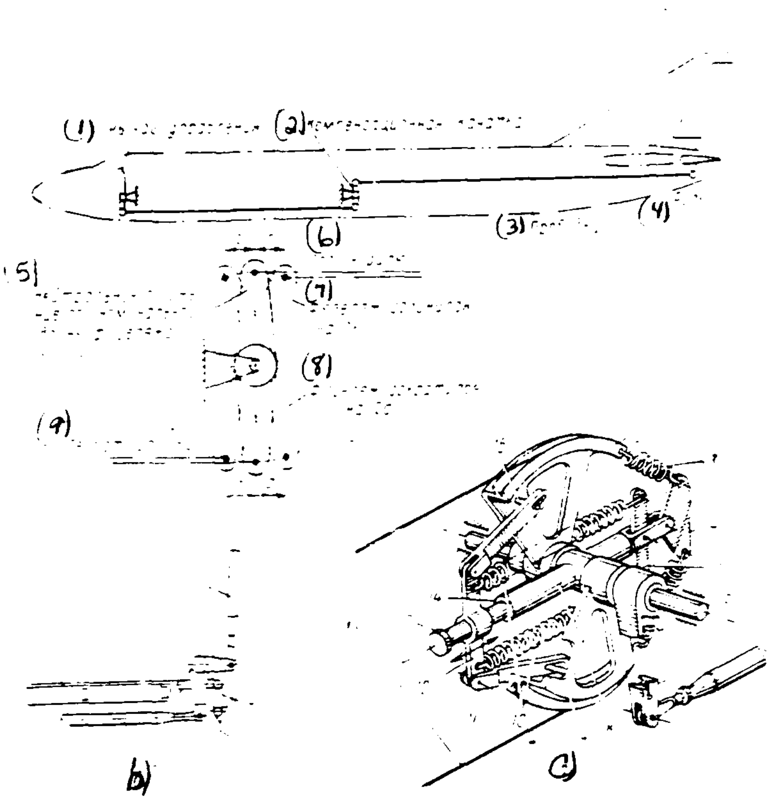

Fig. 19.36. Structural/design measures for compensation for linear structural distortions of aircraft in control line: a) compensating rocker and its installation on aircraft; b) the compensating mechanism: 1 - floor of the cabin/compartment (it is not deformed from the air loads and the overpressure); 2 - control rod (from the pilot); 3 - bottom of passenger pressurized cabin; 4 - compensating mechanism; 5 - rocker; 6 - rod/thrust to the control; c) the mechanism of the tension of cables; 1 - cables; 2 - upper mobile sector; 3 - spring of the tension of cables; 4 - rocker of the alignment/levelling the tensions of upper and lower cables; 5 - rocker; 6 - rotational axis of mechanism, to which motionlessly are fastened lever 7 and yoke/arm 8; 9 - lower mobile sector; 10 - spring of the tension of

cables during the structural distortions of aircraft; 11 - link with the slot; 12 - slider; 13 - stop of slider with the maximum fineness ratio of fuselage; 14 - stop of slider with the maximum decrease of the length of fuselage; 15 - more fingered.

Key: (1). Control lever. (2). Compensating rocker. (3). Guide. (4). Control. (5). Neutral position with nominal length of fuselage. (6). Rod/thrust to control. (7). Fuselage was lengthened to 2b. (8). Fuselage was reduced by 2a. (9). Rod/thrust from pilot.

§ 10. Connection to the guide of the actuating mechanisms of automatic systems.

Difference in designations/purposes of automatic devices/equipment, used in systems of control of contemporary aircraft, leads also to difference in methods of connection of their actuating mechanisms to mechanical guide (see Fig. 19.20a).

High speed actuating mechanisms of systems of increase in stability and damping are connected to guide consecutively/serially so that their work would not be reflected on change in position of control levers and efforts/forces on them. For guaranteeing this it is necessary to establish/install the mechanisms of these systems as close as possible to the input devices of the hydraulic boosters of the drive of the managers of surfaces so that the efforts/forces, created by these mechanisms in the guide and which affect along it into both sides from their site of installation (i.e. to the control lever in pilot and to the output component/link of hydraulic booster), from the side of control lever would be extinguished by the inertness of the elements of the largest possible section of guide.

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Work of actuating mechanism, mounted into guide consecutively/serially, introduces disagreement/mismatch between clauses of lever and control, than, strictly, and is reached possibility of independent work of pilot and this mechanism. Because of this, as shown in the limitations of the deviations of lever and control Fig. 19.37b, dependence of the control displacements δ from the deviation of lever x in not the neutral positions of output motion rod can appear.

Limitations of range of angular deflection of control in the case of possible failures of automatic systems in not neutral (but all the more, in extreme) positions of output sections of their actuating mechanisms it is possible to lead to difficulties in balancing/trimming of aircraft in some flight conditions. The limitation of the range of deflection of control lever due to the premature stop of the intake component/link of hydraulic booster in not the neutral position of the output component/link of actuating mechanism can lead to the onset on this of the mechanism of uncalculated loads from the pilot feel, which yet did not reach its stop.

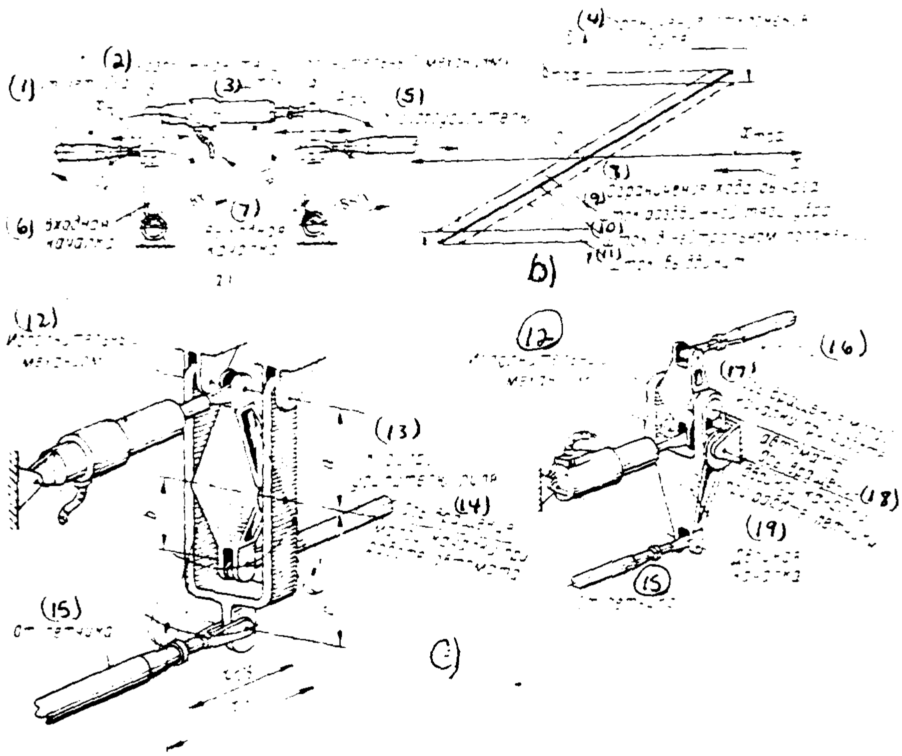

Fig. 19.37. Consecutive connection to guide of actuating mechanisms of automatic systems: a) telescopic rod/thrust; b) change in kinematic dependence of deflections of control with deviations of lever of control in different positions of stock/rod, consecutively/serially connected to guide actuating mechanism (telescopic rod/thrust); c) two examples of consecutive installation of actuating mechanisms with the aid of differential (summing) rocker. Key: (1). From the pilot. (2). Telescopic rod/thrust (actuating mechanism). (3). Stock/rod. (4). Limitation of deflection of control. (5). To hydraulic booster. (6). intake rocker. (7). Output rocker. (8). Limitations of course of lever. (9). Stock/rod of telescopic rod/thrust is retracted. (10). Stock/rod in neutral

position. (11). Stock/rod is advanced. (12). Actuating mechanism. (13). To hydraulic booster of control. (14). Rotational axis of low rocker with work of automatic machine. (15). From pilot. (16). To control. (17). Rotational axis of low rocker with work of automatic machine. (18). Rotational axis of double rocker with work of pilot. (19). Dual rocker.

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These special features of kinematics of control line with series-connected actuating mechanisms of automatic systems must be considered during selection of ranges of deflection of control levers (which are regulated by stops in levers themselves) and of ranges of deflection of controls (which they are regulated by stops in intake components/links of hydraulic boosters).

Actuating mechanisms (control actuators) of systems of trajectory control (STU) and, in particular, autopilots as this is shown in Fig. 19.20a, on majority of contemporary aircraft they are connected to guide in parallel. Therefore with the operation of these systems cockpit controls are moved with respect to the control displacements of aircraft, and the possibility of the simultaneous work of pilot and automatic system is precluded. For guaranteeing the possibility of piloting by pilot during the disconnection of the system of automatic trajectory control (autopilot) the control actuators of these systems have devices of the disconnection of their output components/links, connected with the guide, from the remaining mechanism. In electrical control actuators these are special cam or friction clutches with electrical control, in hydraulic - electrically controlled valves of cross-feed of the cavities of actuating cylinder. Switching on and disconnection of these devices/equipment occurs simultaneously with switching on or disconnection of the system of automatic trajectory control.

On aircraft with fully powered controls this method of connection to guide of actuating mechanisms of systems of automatic trajectory control leads to number of new problems, connected with presence of loaders. On one hand, the power of these mechanisms must be sufficient in order to overcome the efforts/forces, created by loaders. On the other hand, the aircraft and pilot must be warnewd from the rapid deviations of levers and connected with them controls during the disconnection (in particular, emergency) of the system of automatic control, during which operate/wear the devicesof the disconnection of the output components/links of control actuators from their mechanisms. As a result of this under the action of efforts/forces from loaders the levers and the controls connected with them will approach rapidly to return the initial balancing position, in which they were located at the moment of the switching on of the automatic system of piloting.

These problems can be solved, for example, by introduction of automatic trimming of efforts/forces from loaders with the aid of mechanism of trimmer effect during automatic flight control. However, this complicates control system and to a certain degree its reliability is decreased. Therefore at present appeared and also the systems of automatic trajectory control, in which actuating mechanisms are established/installed in the guide consecutively/serially and with their work the cockpit controls remain fixed, as it is made, for example, on aircraft B-58 "Hustler" (USA), etc.

Similar systems also possess number of shortcomings in particular, impossibility of control/check by pilot of work of system on change in position of control levers. The failures of this automatic system in not the neutral position of the output component/link of actuating mechanism will lead to the difficulties described above, which are aggravated by the substantially higher control displacements, required for the system of automatic trajectory control, than for the systems of an increase in the stability and damping.

In connection with problems examined above question about selection of one or the other method of connection to guide of actuating mechanisms of systems of automatic trajectory control for each specific projected/designed aircraft must be solved especially, although it is possible to note that systems, in which control levers with operation of the automatic control systems of flight are moved, seem more logical and corresponding to psychology of pilots.

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Mechanisms of gear reduction $k_{ш}$ from lever to control of aircraft compose special group of actuating mechanisms of automatic systems connected to guide. According to the designation/purpose these mechanisms with the accomplishment of their objective, in contrast to actuating mechanisms examined earlier of the stabilization systems and STU, must not diverge controls. However, if we examine the dependences given in Fig. 19.38 of the control displacement δ from the course of control lever x with different gear ratios, adjusted by the actuating mechanism of the control system of gear ratio in the guide, then one can see well that with gear reduction in not the neutral position of control lever (for example, $x_{(a.n)}$) unavoidably must occur a change in the clause ("tilt") of control to value $\delta_{yв}$. If in this case it is necessary to preserve the initial balancing position of control $\delta_{бал}$, then with the work of the mechanism of gear reduction for pilot it is necessary to change the position of control lever to value $x_{г3}$ for the compensation for the "tilt" of control.

"tilt" of controls with work of mechanisms of gear reduction in guide of path and lateral controls does not cause special difficulties, so these controls and corresponding levers during larger part of flight are located in neutral (or close to it) situation, when, as is evident from Fig. 19.38, gear reduction does not entail change in position of control surface or control lever.

It is a different matter in guide of pitch control. Here, as is known, the positions of control surface and corresponding control lever change in different flight conditions within sufficiently wide limits for guaranteeing balancing/trimming aircraft [in accordance with balancing curves $\delta_B=f(M, H, \bar{x}_T)$]. Therefore in the channel of pitch control the work of the mechanism of gear reduction will lead to noticeable change in one of the most important indices of handling in pitch - balancing curves $x_B=f(M, H, \bar{x}_T)$.

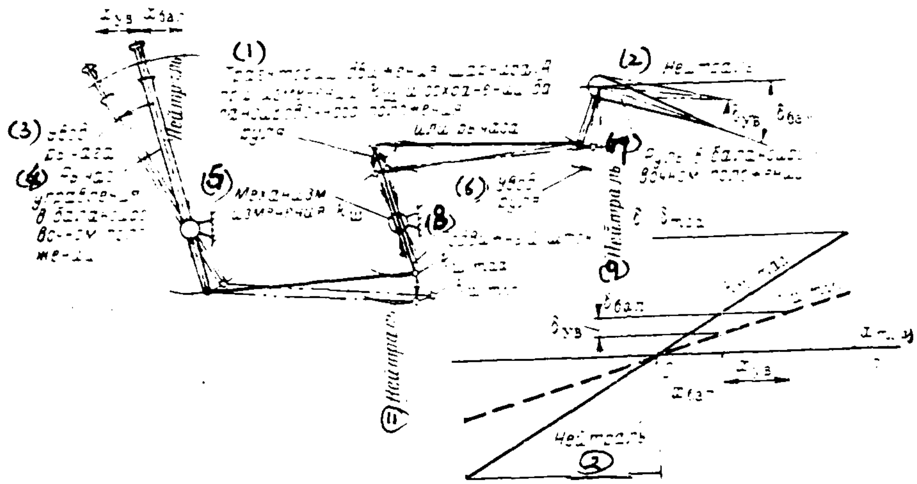

Fig. 19.38. To phenomenon of "tilt" of control (or control lever) with work of automatic machine of gear reduction in control line (diagram is shown for case of work of automatic machine with decrease transmitted number, $\delta_{ув}$ - tilt of control with fixed lever; $x_{ув}$ - deviation of lever, required for compensation for tilt of control (tilt of lever); $k_{ш}=d\delta/dx$

Key: (1). trajectory of the motion of hinge A with the change ... and the retention/maintaining of the balancing surface position or lever. (2). Neutral. (3). Tilt of lever. (4). Control lever in the balancing position. (5). Mechanism of change. (6). Tilt of control. (7). Control in the balancing position. (8). Mobile stock/rod. (9). Neutral.

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It is possible to significantly improve these balancing curves, utilizing the phenomenon of "tilt" for the compensation for the transonic instability of aircraft on the speed and the decreases of "ladles" in the balancing curves. The at the same time unsuccessful installation of the mechanism of gear reduction can lead to an essential deterioration in the balancing curves.

Let us note that effect of mechanism of change in transmission number to change in balancing curves $\varphi_{b}=f(M, H, \bar{x}_{T}, \ldots)$, besides prescribed/assigned law of regulation $k_{\text{ш}}=f(M, H)$, is determined by selection of flight conditions and balancing clause corresponding to it of altitude control, with which rods/thrusts of control line are established/installed at angle of 90° to axis of stock of mechanism. In this position of the mechanism of the motion of stock/rod, with which occurs a change of transmission number in the guide, they do not lead to a change in the clause of control [1].

FOOTNOTE [1]. In more detail a question about the effect of the "tilt" of altitude control on the balancing curves and the use of this phenomenon for their improvement see [20], page 242-248. ENDFOOTNOTES.

With installation of actuating mechanisms of stabilization systems and automatic control and their connection to mechanical guide for guaranteeing assigned gear ratios are made are sufficiently simple kinematic calculations, whose examples we will examine below.

As shown in Fig. 19.37, consecutive connection of actuating mechanisms can be accomplished/realized either by method of their direct inclusion in guide as telescopic rods/thrusts or with the aid of differential (summing) rockers. Usually the design of the units of the connection of the actuating mechanisms of automatic systems is conducted already after the installation of the control levers and hydraulic boosters of the drives of control surfaces. Therefore during the design of these units it must be prescribed/assigned:

1) the complete course of rod/thrust (or cable) from the control lever of pilot x_{π};

2) the complete course of rod/thrust (or cable) to hydraulic booster $x_{r.y.}$ the ensuring complete range of angular deflection of this control surface δ_{max}, moreover sufficiently frequently is assigned $x_{\pi} = x_{r.y.}$;

3) the gear ratio of actuating mechanism μ, expressed by the value of the course of its output component/link in mm by one of a change in that parameter of the flight, which for this system is command (for example, the angular rate of rotation of aircraft for the vibration damper, angle of attack for the automatic machine of stability, etc.);

4) the required gear ratio k in degrees of the deviation of the control surface by one of a change in the same parameter.

Selection of method of consecutive connection of actuating mechanism is defined by both the type of mechanism itself and by the method of fastening provided for by its construction/design and by value and form of available free volumes during design of unit of installation of this mechanism.

Installation of actuating mechanisms by telescopic rod/thrust.

With this method actuating mechanism is included in guide as by one of rods/thrusts and hinged is fastened for two ends/leads between two rockers (see Fig. 19.37a). For guaranteeing the required gear ratio reaches of rockers are selected from the following considerations.

Output arm $l_{вых}$ is selected in such a way that with prescribed/assigned course of rod/thrust to hydraulic booster general/common deviation of rocker would not exceed, as we recommended earlier, ±30-35°.

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If we accept smaller of these values (i.e. ±30°), then the output arm of rocker will be determined: $l_{вых} \geqslant x_{шт}$. From the simple proportion we determine, what course of rod/thrust to hydraulic booster x provides the deviation of control surface of angle, numerically equal to the required gear ratio k:

$$\frac{x_{шт}}{\delta_{max}} = \frac{x_k}{\delta_k}.$$

but hence $x_k = \frac{\delta_k x_{шт}}{\delta_{max}}$, where $\delta_k = k°$. With the aid of the second proportion we determine arm $l_з$ of the connection of actuating mechanism to the output rocker: $\frac{l_з}{l_{вых}} = \frac{\mu}{x_k}$, whence $l_з = \mu \frac{l_{вых}}{x_k}$.

If on conditions of task was prescribed/assigned $x_т = x_у$ then intake rocker will have the same arms, so forth $l_{вх} = l_{вых}$ and $l_Б = l_А$.

But if $x_т \neq x_у$ then reaches of intake rocker are found from relationship/ratio: $\frac{x_т}{x_у}\frac{l_{вых}}{l_{вх}} = \frac{l_А}{l_Б}$ or when $l_A = l_Б$ $l_{вх} = l_{вых}\frac{x_т}{x_у}$. If as a result of calculation unit is unconstructive or badly/poorly is composed, then calculations should be repeated, after assigning another value $l_{вых}$.

Installation of actuating mechanisms with the aid of the differential (summing) rockers.

Together with direct installation of actuating mechanisms of artificial stability installations and STU into control line, for layout reasons can be used diverse variants of installations of these mechanisms with the aid of differential (summing) rockers. Examples of such installations are shown in Fig. 19.37c.

Arms of rockers, which form differential (summing) mechanism, are selected depending on relationship/ratio of three values: command displacement/movement (course) of rod/thrust (or cables), which is adequate/approach site of installation of mechanism from control lever in cockpit $x_т$, course of rod/thrust from this place to hydraulic booster of drive of autopilot surface $x_у$ and course of stock/rod of actuating mechanism. It is natural that the rod/thrust (or cable) with the greatest course is connected to the largest of the available three arms of rockers, which form differential mechanism.

Kinematic calculation of this mechanism analogous with that given above is reduced to solution of series/row of proportions and finding of reaches of rockers, which ensure required gear ratio of automatic system k and maximum angle of deflection control δ_{max}.

Let us examine example of calculation of mechanism, shown in Fig. 19.37c, to the left. This type of differential mechanism is conveniently used, if on the conditions of task $x_{п} > x_{гп}$ (if on the conditions of task prescribed/assigned $x_{п} < x_{гп}$, then the attachment points of rod/thrust from the pilot and the rod/thrust to the hydraulic booster should be changed by places).

Analogous with example dismantled/selected above of calculation of installation of telescopic rod/thrust let us first determine course of rod/thrust of guide to hydraulic booster δ_k, which ensures control displacement of angle x_k, numerically equal to required gear ratio $k: \frac{x_{гу}}{\delta_{max}} = \frac{x_k}{\delta_k}$, whence $x_k = \delta_k \frac{x_{гу}}{\delta_{max}}$, where $\delta_k = k$. After this, of the design considerations we are assigned by the value of less of three arms of the rockers of mechanism (a, b or c in Fig. 19.37c): let, for example, $b=B_1$. Then we can compose the proportions: $\frac{u}{a} = \frac{x}{B_1}$ and $\frac{x}{a-c} = \frac{x_{гу}}{a-B_1}$, from which easily are located not the known arms of the rockers of the differential mechanism a and c.

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If as a result of the first kinematic calculation are obtained such values of the arms of rockers a, b and c, with which mechanism is difficult to realized structurally or badly/poorly it is composed on the aircraft, then calculation one should repeat, after assigning another value of $b=B_2$.

Analogous calculations are performed also during application of differential mechanisms of other forms.

Parallel connection to the guide of the actuating mechanisms (control actuators) of the systems of trajectory control (STU).

Depending on type of these mechanisms their connection to guide is accomplished/realized either with the aid of double- armed rockers (if output component/link of control actuator has forward movements, Fig. 19.39a), or with the aid of rocker with sector (if control actuator it has output component/link, carried out in the form of drum or chain wheel, that accomplish reversible rotary motions with work of machine, Fig. .19.39, b, c).

Given gear ratios in these installations are determined as a result of corresponding kinematic calculation of reaches of double-armed rockers and radii of sectors of sectors. This calculation does not present difficulties and analogous with examples examined above it is reduced to the solution of proportions.

However, with parallel connection of actuating mechanisms in contrast to consecutive is necessary to provide even conformity of maximum courses of rods/thrusts or cables of control line to maximally possible courses of output components/links of these mechanisms so that during disconnection of control actuator of limitation of course of output component/link control actuator they would not lead to limitation of course of control line.

With sufficiently large maximum course of cables from output component/link (drum) of control actuator of rotary action for its connection to control line is necessary to use two-stage transmission as this shown in Fig. 19.39c, to the right.

At conclusion of present paragraph let us note that for contemporary aircraft construction tendency toward use of combined irreversible hydraulic boosters instead of special control actuators of systems of trajectory control (STU) and autopilots is characteristic. The combined hydraulic boosters have two groups of the distribution valves, of which the pilot with the aid of the mechanical guide, controls one, and another - computing blocks of the automatic stabilization systems and trajectory control with the aid of the electrical signals.

§ 11. Selection and the installation of the hydraulic boosters of the drive of control surfaces.

Correct selection and installation of hydraulic boosters of drive of controls is one of most important tasks of designing entire complex of control of contemporary aircraft.

For contemporary high-speed/high-velocity (transonic and supersonic) aircraft characteristically ever wider application of irreversible stick-and-power control circuits without possibility of transition/transfer to emergency manual control with the aid of only muscular force of pilot. Application of such a control gives a whole series of the advantages, from which should be indicated the following:

1) the possibility of applying the controls without aerodynamic balance, which makes it possible to decrease their areas due to an increase in the effectiveness; at the same time this installation because of decreasing of resistance raises aircraft quality/fineness ratio (by 0.5-0.8);

2) the possibility of guaranteeing the necessary flutter characteristics of control surfaces without their mass balancing, and only for the account of use of the rigid and damping characteristics of hydraulic boosters (on aircraft VC-10, for example, this it made it possible to remove/take about 450 kg of weight compensators of controls);

3) the possibility of simplification in the construction/design of hydraulic boosters (and consequently, increase in their reliability) as a result of the withdrawal of a whole series of the devices/equipment, which provided transition/transfer to manual (boosterless) control.

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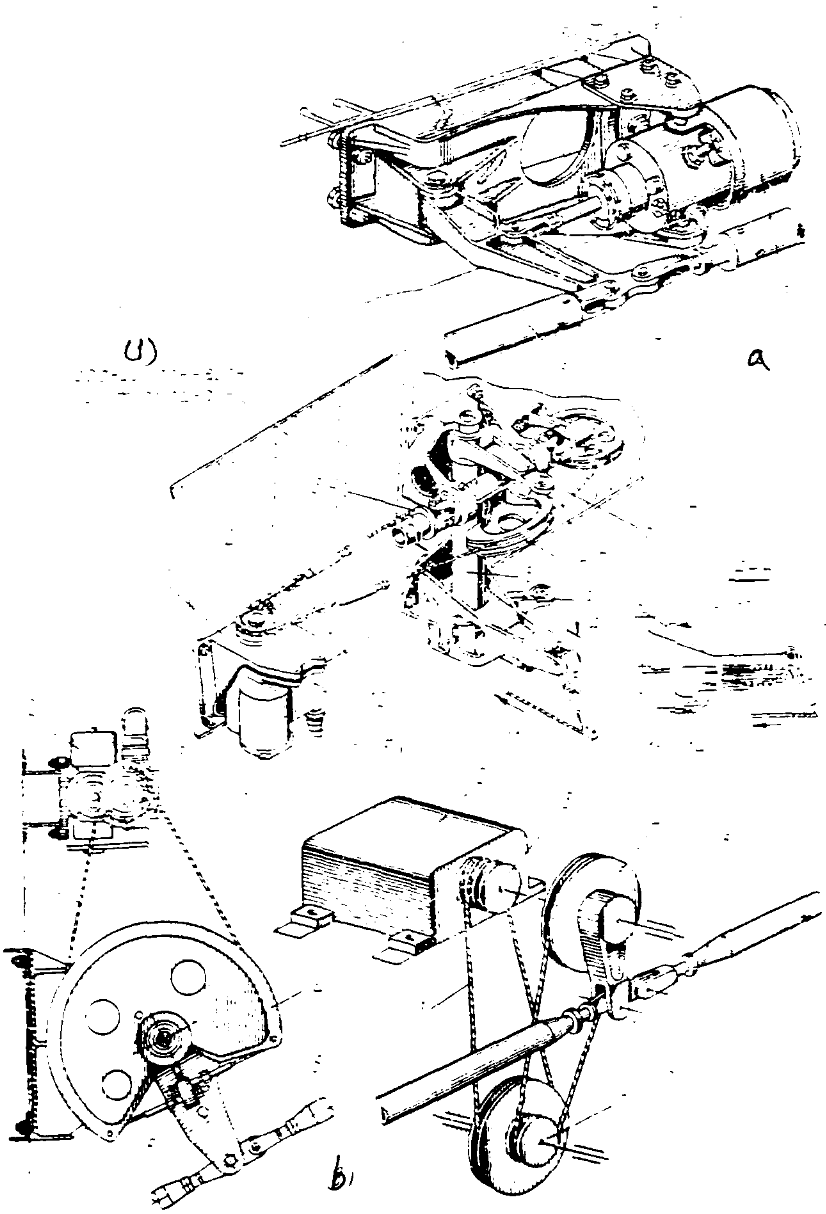

Fig. 19.39. Parallel connection to guide of actuating mechanisms of systems of trajectory control (control actuators of autopilots): a) connection of control actuator with forward motions of output component/link (stock/rod): 1 - cantilever; 2 - removable bracket-jaw

of fastening the pin/journal of control actuator; 3 - control actuator; 4 - rocker; 5 - rod/thrust of control line; b) an example of the installation of control actuator with rotary motion of output component/link (chain wheel) and of control assembly of ailerons on rear longeron of the wing center section: 1 - control actuator; 2 - chain wheel; 3 - drive shaft of flaps; 4 - roller; 5 - lever of the drive of the control line by ailerons in the wing; 6 - cable; 7 - sector of the connection of control actuator; 8 - shaft of the translation of motion of control by ailerons; 9 - lever of the connection of rod/thrust from the cockpit (from the handwheel); 10 - lever of the system of the parking locking of ailerons; 11 - cable from the lever of the locking of ailerons in the cabin/compartment (by arrow/pointer shown the direction of the motion of cable with the locking); 12 - circuit; 13 - catch; 14 - seat of catch into the lever; 9; 15 - the radial bearing: c) two examples of connection to the guide of control actuators with rotary motion of output component/link - cable drum (single step transmission to the left and two-stage - to the right): 1 - cable; 2 - control actuator; 3 -cable drum; 4 - sector with the rocker; 5 - control line; 6 - connection cable in the sector; 7 - double pulley sector of the second step/stage.
Key: (1). Rear longeron/spar of wing center section.

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However, application of irreversible stick-and- power control circuits without transition/transfer to emergency manual control advances sufficiently serious problems of providing flight safety and completion of fitting/landing aircraft in the case of possible rejection in this system. Some methods of resolution of these problems will be examined in the following paragraph.

So that contemporary hydraulic boosters could promptly accomplish task in guarantee of flutter characteristics of control surfaces, in their construction/design series/row of improvements, which raise their dynamic rigidity (application of so-called "inverted" kinematic diagram, in which direction of motion of intake and output components/links of hydraulic booster they are opposite, introduction of supplementary feedback) is used. At the same time especially serious attention must be focused on the guarantee of maximum rigidity of the points of attachment of hydraulic boosters to the construction/design of aircraft, and also rigidity of the section of guide from the hydraulic booster to the autopilot surface.

It was above noted that for contemporary aircraft construction is characteristic tendency to completely remove this powering unit of guide and to fasten output component/link of hydraulic booster directly for lever of drive of control surface. Besides an improvement in flutter characteristics due to an increase in the dynamic rigidity of the drive of control surface (depending, in particular, on the rigidity of guide from the control to the hydraulic booster), this mounting method of hydraulic boosters provides a minimum quantity of elements of guide, stressed from the hinge moment of autopilot surface, and, consequently, minimum weight of guide.

Let us note that during installation of hydraulic boosters directly on control surfaces (especially in wing and tail assembly) always one ought not to attempt "to hide" hydraulic booster and lever of drive of control within theoretical enclosure. The tendency "to hide" the drive of control surface into the enclosure, as a rule leads to the complicated and heavy constructions/designs, with the possessing, furthermore, an even more considerable friction and with the gaps, which worsen/impair the quality of control system.

Considerably simpler and by lungs, and main thing, more rigid, are constructions/designs, in which axis/axle of hydraulic booster is established/installed perpendicularly to rotational axis of control surface. The stationary part of the hydraulic booster in this case is fastened for sufficiently rigid wing center section or tail assembly, and its output mobile link - directly for the lever of the drive of control surface, whose length in this case can be selected the optimum, i.e., ensuring minimum weight construction/design of drive. hydraulic booster and lever of the drive of control surface with this mounting method is closed by the non-load-bearing fairings (Fig. 19.40) appearing in the flow. The constructions/designs, analogous to that shown in Fig. 19.40, extensively are used on the contemporary aircraft both on the subsonic and on the supersonic, the decrease of the profile thicknesses of wings and tail assembly of which leads to the practical impossibility "to hide" the drive of control inside the enclosure.

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It is natural that any flanges in theoretical enclosure of surfaces of aircraft streamlined with flow increase its resistance. However, as a result of the smallness of the midsection of the fairings about the drives of controls in comparison with the midsection of aircraft, this increase in the resistance due to the fairings appearing into the flow is very insignificant and, unconditionally, it is redeemed by considerable gain in the weight, rigidity and simplicity of the construction/design of drive.

During design of drives of completely rotatable control surfaces (for example, controllable completely rotatable stabilizer or of fin) it is also necessary to attempt to establish/install hydraulic boosters in such a way as to fasten their output components/links directly behind lever of drive of these surfaces. Tendency to improve flutter characteristics of the completely rotatable controllable stabilizer leads to the need of applying the separate hydraulic boosters for the drive of its right and left of halves. Only in the case of applying common straight axle, structurally connected with both half of stabilizer, the application of one hydraulic booster is possible, if it is possible to ensure a sufficient rigidity of comparatively large sections of the tubular of workers for the torsion.

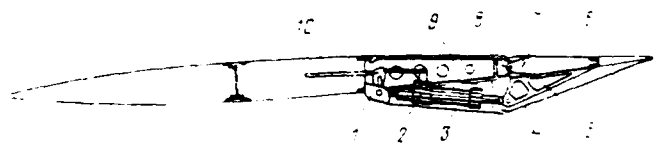

Fig. 19.40. example of installation of hydraulic booster of drive of control surface on thin wing of supersonic aircraft: 1 - mounting bracket of hydraulic booster; 2 - hydraulic booster; 3 - stationary part of fairing; 4 - moving element of fairing; 5 - lever of drive of control; 6 - control; 7 - longeron of control with bracket of charge; 8 - rear longeron/spar (wall) of wing; 9 - supporting rib of wing; 10 - control rod hydraulic booster.

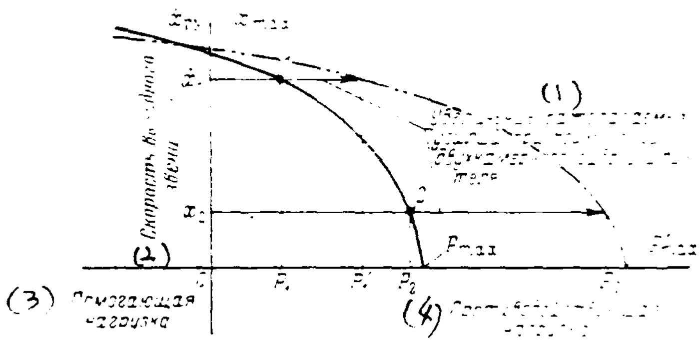

Fig. 19.41. External characteristic of one-dimensional hydraulic booster (———) and its change with twofold redundancy of available power by applying two-chamber hydraulic booster with feeding of chambers/cameras from two independent hydraulic systems (—··—··—). Points with the numerals in the unbroken curve correspond to two rated flight performances, on which is determined the required characteristic of the hydraulic booster: 1 - light loads, high speeds reversals; 2 - large loads, low speeds reversals.

Key: (1). increase in the available efforts/forces during the application of two-chamber hydraulic booster. (2). Speed of output component/link. (3). Helping load. (4). reactive load.

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Selection of hydraulic boosters for drive of control surfaces of aircraft is produced on the basis of obtained values of required speeds reversals (deviations) of these surfaces and their hinge moments. In this case is utilized the known dependence of the speed of the motion of output link of hydraulic booster on the effort/force, applied to its output component/link, i.e., its external (mechanical) characteristic (see Fig. 19.41), described by the equation

$$\dot{x}_{\text{г}} = k_x (P_{max} - P). \qquad (19.1)$$

where $\dot{x}_{\text{г}} = dx_{\text{г}}/dt$ - speed of the motion of the output component/link of hydraulic booster in mm/s;

k - proportionality factor;

P - the instantaneous value of the effort/force, applied to the output component/link of hydraulic booster;

P_{max} - maximum value of the reactive effort/force, with which the speed of the motion of the output component/link of hydraulic booster becomes zero.

Value P_{max} for hydraulic booster is determined by area of its piston F and by maximally possible drop/jump in pressures ΔP_{max} in working cavities $P_{max} = F\Delta p_{max}$.

For convenience and simplification in calculations expression

(19.11) for external characteristic of hydraulic booster is utilized into somewhat another form, which points out dependence of possible angular velocity reversals of control δ on hinge moment $M_{ш}$:

$$\dot{\delta}^2 = k(M_{ш\,max} - M_{ш}). \qquad (19.12)$$

where $M_{ш\,max}$ - value of hinge moment, which is created on output component/link of hydraulic booster of stress P_{max}.

In order to select hydraulic booster, which ensures the necessary for speed reversals of control in all flight conditions, first they are utilized the data about the necessary speeds reversals and the corresponding hinge moments in two flight conditions:

1) flight conditions, where are necessary maximum speeds reversals $\dot{\delta}_1$ with comparatively small values of hinge moments $M_{ш}$ (usually this takeoff and landing regimes);

2) flight conditions, where on the controls appear greatest hinge moments $M_{ш}$, but required speed reversals $\dot{\delta}_2$ - is comparatively small.

Equation (19.12), written for these two flight conditions, gives two equalities, where they remain unknowns k and $M_{ш\,max}$:

$$\dot{\delta}_1^2 = k(M_{ш\,max} - M_{ш1}); \quad \dot{\delta}_2^2 = k(M_{ш\,max} - M_{ш2}).$$

After subdividing these equalities one to another, we will obtain $\frac{(\dot{\delta}_1)^2}{(\dot{\delta}_2)^2} = \frac{M_{ш\,max} - M_{ш1}}{M_{ш\,max} - M_{ш2}}$, whence is easy to determine value $M_{ш\,max}$, and then from any equality and the value of coefficient k.

Obtained values $M_{ш\,max}$ and k make it possible to construct required external characteristic of hydraulic booster, written in the form (19.12), and to test guarantee of inequality $M_{\text{ш потр}} \leqslant M_{\text{ш расп}}$ and $\dot{\delta}_{\text{потр}} \leqslant \dot{\delta}_{\text{расп}}$ in all flight conditions. Then, utilizing relationships/ratios $\dot{\delta} = \frac{\dot{x}_{\text{гу}}}{R\cos\delta}$ and $M_{\text{ш}} = PR\cos\delta$ (where R - length of the lever of the drive of control surface, and δ - angle of its deviation from neutral position, Fig. 19.42), it is possible to obtain required external characteristic of hydraulic booster in the form (19.11), on which either is selected finished hydraulic booster, or orders herself new specialized OKB.

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Let us note here that required power of hydraulic booster ($N_{\text{гу п тр}}$) does not depend on length R of lever of drive of control surface and is determined by expression

$$N_{\text{гу п тр}} = \frac{M_{\text{ш max}} \dot{\delta}_{\text{max}}}{57.3} \text{ кгс·м/с} \qquad (19.13)$$

($M_{\text{ш max}}$ it is expressed in kgf•m, and $\dot{\delta}_{\text{max}}$ - in deg/s).

Required power of hydraulic booster calculated according to formula (19.13) at prescribed/assigned design pressure p in hydraulic system determines required fluid flow rate through this hydraulic booster during motion of its output component/link $Q_{\text{гу п тр}} = N_{\text{гу п тр}} \, 0.167 \, p$ l/min ($N_{\text{гу п тр}}$ it is expressed in kgf•m/s, and p - in kg/cm²), and total required expenditure for all n hydraulic boosters of control system it is determined by sum $Q_{\text{п тр}} = \sum_{i=1}^{n} Q_{\text{гу п тр}}$. This sum determines the required power of hydraulic system for the feeding of the hydraulic boosters of the drives of control surfaces. If into this hydraulic system are connected even and other users, then the general/common required capacity of the pumps of this hydraulic system will be determined by the sum of the expenditures/consumptions of all users taking into account the diversity factor of their work [1].

FOOTNOTE [1]. During the determination of the required power of the hydraulic system of the feeding of hydraulic boosters (by the so-called booster) the diversity factor of the work of users is taken as equal to 1, since all drives of control surfaces in contrast to other users of hydroelectric energy can work simultaneously. ENDFOOTNOTES.

With the advent of and development of stick-and-power control circuits power of aircraft hydraulics began extremely rapidly to grow/rise, although required powers for sulfonation of other users (for example, retraction and landing gear lowering and high-lift device of wing, drive of mechanism of turn of front/leading strut, etc.) virtually little grew/rose. Thus, for example, according to the data of journal Aircraft Engineering (No 12, 1966) the power of the hydraulic systems of the main-line passenger aircraft DC-7 (boosterless control). VC.10 and "Concorde" compose with respect 34, 160 and 450 hp.

Independence of required power for deviating control surface from length R of lever of its drive does not indicate, however that arbitrary selection of this length is feasible. With too low values R the quality of actuator deteriorates by the fact that the gaps in hinge connections become sufficiently noticeable against the background of comparatively low required command displacements/movements, in consequence of which the angle of the free deviation of control surface increases. At the same time the efforts/forces from the hinge moment, which affect on hydraulic booster and hinges of its fastening to the lever and the construction/design strongly grow/rise, which raises the weight of these elements. And although with the low required command courses of the output component/link of booster its sizes/dimensions and, consequently, weight decrease, the total weight of unit can prove to be sufficiently to large.

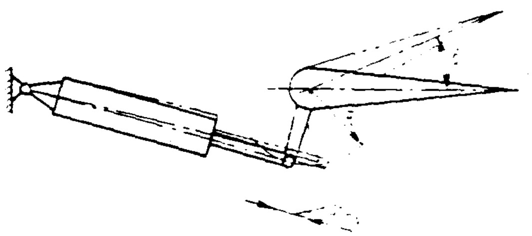

Fig. 19.42. Connection/communication of course of output component/link of hydraulic booster with angle of deflection of control surface δ and length of lever of its drive R.

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With too great values R weight of structure of aircraft as a result of decrease of effective efforts/forces is reduced; however due to considerable increase of weight of hydraulic booster itself with large required course of its output component/link total weight of unit also can increase. Let us note that an increase in the course of the output component/link of the hydraulic booster (more precise - the course of its piston) much more essentially it affects the increase of the weight of hydraulic booster itself, than the increase in the effort/force, which it must receive. This explains those that in the first case, especially if hydraulic booster two-chamber, strongly grows/rises its length, and an increase in the received effort/force is achieved by a very small increase in the diameter of actuating cylinder. Furthermore, and with sufficiently high values of R the diameter of this cylinder frequently is necessary additionally to increase (in comparison with the required for the perception of the effective efforts/forces) for increasing the damping properties of hydraulic booster and guarantee of the necessary flutter characteristics of autopilot surface.

It is obvious that for each specific conditions there is some optimum length of lever of drive of autopilot surface, with whom total weight of elements of construction/design of aircraft and hydraulic booster is smallest. Task of designer - to find this length.

During arrangement of aircraft and design of units drive of control surfaces must be considered, that contemporary multichamber hydraulic boosters have sufficiently large length. Thus, for example, two-chamber hydraulic boosters with the working stroke of stock/rod 150-200 mm have the adjusting length of order 1000-1200 mm.

During design of charge and drive of control surfaces considerable attention should be focused on selection of sweepback their rotational axes, since from this to a considerable degree required driving power depends. Especially this relates to the completely rotatable swept surfaces, for example, to the controllable stabilizer. On one hand, by an increase of sweepback axis/axle it is possible to attain the decrease of hinged moment with respect to this axis/axle. But, on the other hand, with an increase in the sweepback of the rotational axis of control surface the flow (i.e. comparatively of the incident flow) angles of deflection of this surface φ during the rotation of axis/axle of the same angles δ

decrease. consequently, for guaranteeing the control surface necessary to effectiveness with an increase in the sweep angle of axis/axle it is necessary to increase the angles of rotation of axis/axle δ, and it means, and the speeds of the rotation of this axis/axle $\dot{\delta}$. As it follows from expression (19.13), an increase $\dot{\delta}$ will lead to an increase in the required driving power and, consequently, to an increase in power and weight of its feeding hydraulic system.

It is necessary to pay attention, that into expressions (19.12) and (19.13) symbol $\dot{\delta}$ designate angular velocity reversals of control relative to its rotational axis. Relatively this 1 of axis/axle is determined and is assigned at the design crews also the hinge moment of this organ/control. The connection/communication of the flow angles of deflection of control φ, which determine its effectiveness (for example, for the pitch control - change in the coefficient of pitching moment at the unit angle of the deviation of controllable stabilizer $m_z^{\varphi_в} = \partial m_z/\partial \varphi_в$) by angles of the rotation of its axis/axle δ is described by the expression

$$\operatorname{tg} \varphi \approx \operatorname{tg} \delta \cos \chi_{oc}, \quad (19.14)$$

where χ_{oc} - sweepback of rotational axis.

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As it follows from expression (19.14), with increase in sweepback of rotational axis difference between axial δ and flow φ angles rapidly grows/rises.

Thus, with increase in sweepback of axis/axle χ of control required driving power due to decrease of hinge moment $M_{ш}$ decreases, and due to increase in required angular velocity of rotation of axis/axle $\dot{\delta}$ - it increases (Fig. 19.43). In this figure

$$\bar{\dot{\delta}}=\frac{(\dot{\delta})_{\chi>0}}{(\dot{\delta})_{\chi=0}};\quad \overline{M}_{ш}=\frac{(M_{ш})_{\chi>0}}{(M_{ш})_{\chi=0}};\quad \overline{N}_{п.тр}=\frac{(N_{п.тр})_{\chi>0}}{(N_{п.тр})_{\chi=0}}.$$

For computed values of hinge moments larger from their subsonic and supersonic values were accepted. Obviously, that there is some optimum sweepback of rotational axis (χ_{opt}) with which the required driving power and its feeding hydraulic system will prove to be minimum.

Since required driving power of control surfaces and hydraulic systems of their feeding in contemporary supersonic aircraft are very great, from application of optimum sweepback of axis/axle (at which this power proves to be smallest) it makes it possible to reduce weight of these elements, and means, and takeoff weight of aircraft.

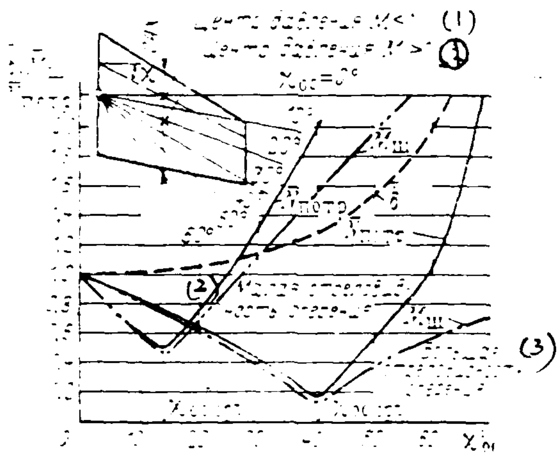

Fig. 19.43. Character of change in relative values of required speed reversals $\bar{\delta}$, hinge moment $\bar{M}_{ш}$ and required driving power $\bar{N}_{потр}$ of completely rotatable empennage with change in sweepback of its rotational axis $\chi_{о}$.

Key: (1). Center of pressure. (2). Low sweepback of tail assembly. (3). Large sweepback of tail assembly.

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§ 12. Measures for an increase in the flight safety during the design of the system of control of aircraft.

During design of boosterless systems of control of low-speed aircraft of control gauges for increase reliability guide was its design with sufficiently large safety factor, and also arrangement/position of guide in such a way, that probability of entry/incidence into it of foreign objects would be smallest.

FOOTNOTE [1]. For example, in flying practice are known tragically the cases, caused by entry/incidence to the guide of the foreign objects, forgotten in the cabin/compartment after conducting of any works (screwdrivers, keys/wrenches, etc.). In order to exclude the possibility of such cases, at present one should insulate the section, where passes guide, from the remaining cabin/compartment. If for layout reasons guide passes above the sealed/pressurized half of cabin/compartment, then from the remaining cabin/compartment it must be isolated by the non-load-bearing (false) floor, openings/apertures in which must be thoroughly sealed, and the places of the passage through it of control levers - are closed with special soft covers. ENDFOOTNOTES.

Together with these measures the separator of guide in the places, easily attainable for the survey/coverage, checkings and conducting of different routine maintenance work is one of the methods of increasing the reliability of control system. This makes it possible to easily reveal/detect the appearing defects and to in proper time remove them.

Enumerated measures provide sufficient reliability of guide also, in stick-and-power control circuits. However, in these systems the degree of their reliability is determined in essence by the degree of reliability of hydraulic booster and hydraulic system of its feeding. Therefore in the first stick-and-power control circuits, in which the hydraulic boosters were still insufficiently reliable, was provided for the possibility of transition/transfer to the emergency manual control with the aid of muscular force of pilot. However, this method of providing safety caused the need of retaining/maintaining the aerodynamic and mass balancing for control surfaces, aerodynamic trim tabs, and also, as a rule limited the possibility of flight in some regimes. At the same time, during the application of a version of transition/transfer to the emergency manual control in hydraulic boosters themselves must be provided for the series of devices (as, for example, the device/equipment of the locking of valve, device/equipment of the guarantee of shock-free transition/transfer, etc.), in turn, that decrease the reliability of hydraulic booster itself.

During use for pitch control of completely rotatable stabilizer contingency transfer to manual control is impossible not only in consequence of very large hinge moments, but also mainly because at subsonic speeds stabilizer, as a rule proves to be overcompensated (as a result of center-of-pressure travel forward from rotational axis).

During use of irreversible stick-and-power control circuit without possibility of transition/transfer in emergency case to manual control only with the aid of muscular force of pilot flight safety, besides other measures for increase in reliability of hydraulic boosters and hydraulic systems of their feeding, is provided by redundancy of these hydraulic boosters and hydraulic systems. At present the principle of the redundancy of hydraulic boosters is most frequently accomplished/realized by an application of the so-called multichamber (two- and three-chamber) hydraulic boosters, which have two or three isolated/insulated chambers/cameras (cylinder), in which are moved the pistons, connected by one common rod. Each chamber/camera of this hydraulic booster feeds by the independent hydraulic system, and piston stroke in it is regulated by the independent distribution valve.

With normal work of multichamber hydraulic booster forces of pressure of liquid on piston in each chamber/camera store/add up on common rod; therefore external characteristic of multichamber hydraulic booster in comparison with single-chamber changes in direction of increase in overcome counter force at the same speed of motion of output component/link of huydraulic booster.

In case of wedging of one of distribution valves of multichamber hydraulic booster they are connected between themselves through decouplers (most frequently spring rods or torsion shafts, workers for torsion), which provide possibility (during application of certain supplementary effort/force) of displacing remaining valves and, consequently, control capability hydraulic booster. Let us note that the pressurization in the chamber/camera with the wedged valve in this case must be disconnected on the signal of the reduction of decoupler.

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In the case of failure of one of independent hydraulic systems of feeding of multichamber hydraulic booster pressure in its appropriate chamber/camera falls and, as is evident from Fig. 19.41, effort/force, which it can perceive hydraulic booster, decrease and, consequently, decrease received (available) hinge moments. This leads to the need for an increase in the power of hydraulic boosters with the redundancy so that even with one working chamber/camera (one operable hydraulic system of feeding) the available hinge moments would exceed required. Otherwise, in the case of failure of one from the hydraulic systems of the feeding of multichamber hydraulic booster, the control surface will spontaneously diverge under the action of hinged moment with all escaping/ensuing from this consequences.

Therefore during application of multichamber hydraulic boosters calculation of required powers, which was being examined in preceding/previous paragraph, is produced for one chamber/camera of hydraulic booster and one hydraulic system, and all remaining chambers/cameras and their feeding hydraulic systems (number of which is equal to permissible number of failures accepted during design during flight) create excess horsepower of hydraulic booster. Value of this excessive (standby) power and, consequently, the excess weight of system is determined by the degree of redundancy and by its diagram.

Let us explain this based on example. Assume it is necessary to ensure continuation of the flight of aircraft and completion of fitting/landing after two failures. This task it is possible to solve by two methods: either for the drive of each control surface to supply one three-chamber hydraulic booster with by striving the independent hydraulic systems of feeding or to supply two two-chamber hydraulic boosters with four independent hydraulic systems of feeding.

In first case total power of three hydraulic systems will be equal to triple cellar, the secondly - total power of two hydraulic systems must be equal only of doubled required, since each of four hydraulic systems can have only half of required power (after two failures it will remain exactly required power). In each specific case depending on required power one or the other solution, which ensures the smallest expenditures of weight for the construction/design of hydraulic systems and hydraulic boosters, can prove to be correct.

On small aircraft of military designation/purpose, as this follows from communications/reports to foreign periodicals, for drive of surfaces of control, as a rule two-chamber hydraulic boosters with feeding from two independent hydraulic systems are used. Also provides supply of one of the chambers/cameras from the standby hydraulics, whose pumps are given not only from the engines of aircraft, but also from any another independent of the energy source. This diagram is provided for for guaranteeing the control capability in the case of the engine failure.

In the case of assumption of limitation of maneuverability of aircraft in the case of failure of one hydraulic system (under condition of guarantee of possibility of completion of fitting/landing) with redundancy of hydraulic systems of feeding of chambers/cameras of two-chamber hydraulic booster it is possible not to provide for redundancy of power (i.e. its increase in comparison with required) described above, if only construction/design of hydraulic boosters provides in the case of failure retention/maintaining available hinge moments and decrease only of available power reversals of controls.

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This condition sufficiently easily is satisfied during application of hydraulic boosters of rotary action, whose two hydraulic motors with feeding from independent hydraulic systems have output through differential to one general/common shaft. With the failure of one hydraulic system and the increase of load on the remaining hydraulic motor its regulator turns inclined washer in the position, in which the speed of motor decreases, and the available moment/torque on the shaft increases.

Such characteristics of two-chamber hydraulic booster of progressive/forward action can be ensured, for example, if pressure from two independent hydraulic systems is conducted through valves of switching to slide-valve distributors of both chambers/cameras. In this case one system must be fundamental for one of the chambers/cameras and emergency for another chamber/camera; another system - fundamental for the second chamber/camera and emergency for the first (Fig. 19.44). In this case in the case of the failure of one from the systems the pressure in both chambers/cameras will be created by the remaining hydraulic system and the available hinge moments in this case will not change. Decrease only the available power the reversals of control, since the capacity of the pumps of one hydraulic system it will be already insufficient for displacing the pistons in two chambers/cameras with the same speed.

Reduction in reliability of hydraulic booster itself due to supplementary valves of switching is shortcomings in this system.

On contemporary heavy aircraft with fully powered controls for increasing safety of flight by case of possible failures is used separation of control surfaces in section, everyone of which is given by independent hydraulic boosters. The feeding of these hydraulic boosters of also realized from the independent hydraulic systems, whose number of the considerations of an increase in the safety reaches four (for example, on the aircraft of North American XB-70A, Boeing-747) and even is more [1].

FOOTNOTE [1]. On the military transport aircraft Lockheed C-5A (USA) is utilized six independent hydraulic systems. ENDFOOTNOTES.

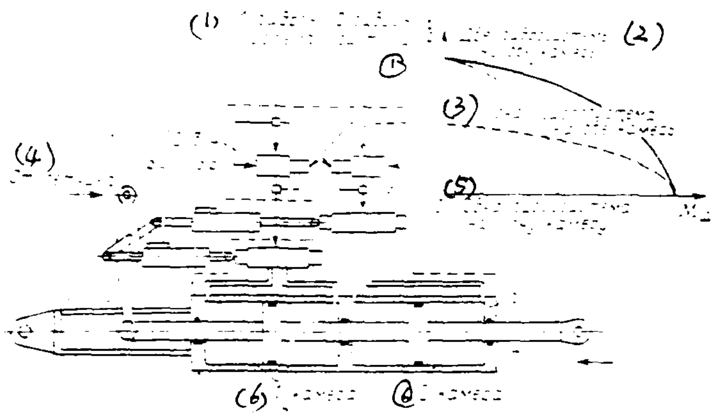

Fig. 19.44. Washing of chambers/cameras of two-chamber hydraulic booster ("inverted" diagram) from two hydraulic systems [(are above to the right shown external characteristics of hydraulic booster with normal operation of two hydraulic systems (solid line) and in the case of failure of one of hydraulic systems (dotted line). Is there for the comparison given external characteristic of one chamber/camera (dot-and-dash line]: 1 - intake lever; 2 - untying spring rods; 3 - all-electric signalling of the wedging of valve; 4 - valves of switching hydraulic systems; 5 - slide-valve distributors; 6 - the actuating cylinder of hydraulic booster; 7 - stock/rod with two pistons (by arrow/pointer shown the direction of the motion of stock/rod during the prescribed/assigned bias/displacement of intake lever).

Key: (1). hydraulic system. (2). Two hydraulic systems into two chambers/cameras. (3). One hydraulic system into two chambers/cameras. (4). From pilot. (5). One hydraulic system into one chamber/camera. (6). chamber/camera.

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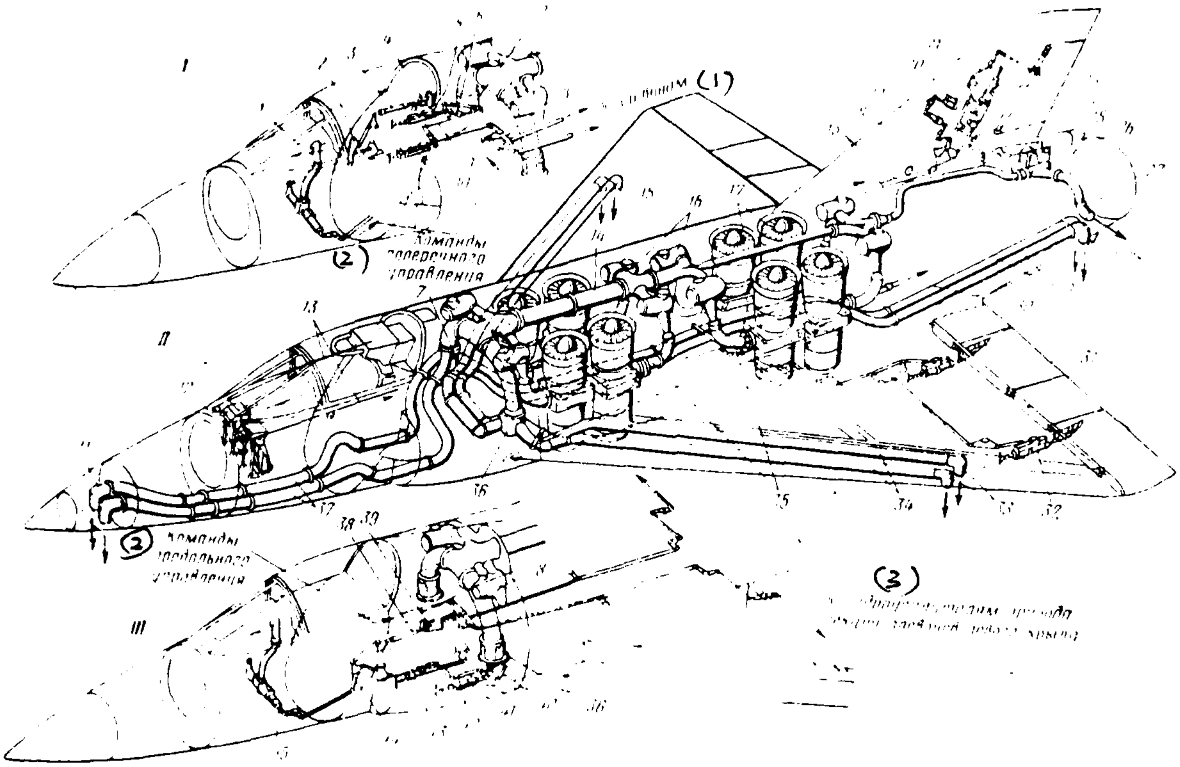

Fig. 19.45. Layout of assemblies of administration on French VTOL aircraft of Dassault "Balzak": I - placement of assemblies of mechanical feature of system of pitch control in forward fuselage: 1 - knob/stick of longitudinal and lateral control; 2 - three-leg rocker of the connection of load system; 3 - loading device; 4 - electrical mechanism of trimmer effect; 5 - electrohydraulic servodrive of control of the valve of the stability-guidance jets of pitch control from the artificial stability installation; 6 - pull control by valve; 7 - valve of the stability-guidance jets of pitch; 8 - mixing mechanism of the commands/crews of the longitudinal and lateral controls of elevons; 9 - broken component/link, which ensures the

possibility of transition/transfer from the manual to automatic control of valve 7 from the stabilization system; 10 - auxiliary hydraulic booster of pitch control; II - arrangement/position of the assemblies of air jet nozzle control and of mechanical feature of control system by rudder; 11 - arrangement of assemblies of stream and mechanical part of system control by rudder direction; 11 - front back up by jet stream control by pitch; 12 - pedal rudder direction controls and by valve of stream by yaw; 13 - cable run from the pedals; 14 - air collector/receptacle of the system of air jet nozzle control; 15 - duplicated/backed up air steering jets along the bank on by right/law wing; 16 - filters of selected/taken from the engines air; 17 - hoisting TRD (8 pieces); 18 - three-leg sector of the connection of the system of the load of pedals; 19 - the loading device of pedals; 20 - mechanism of trimmer effect; 21 - connection of the mechanical control line by the valve of the stability- guidance jets of yaw; 22 - rudder; 23 - hydraulic booster; 24 - lever with the variable arm for regulating the gear ratio; 25 - cable control line by the valve of the stability-guidance jets of yaw; 26 - valve stability-guidance jets of yaw; 27 - air nozzle (left) of jet-edge yaw steering; 28 - rear duplicated/backed up air jets pitch control; 29 - back up air ducts to nozzles 28; 30 - air bleed to the turbopump unit; 31 - two sections of elevons (left wing); 32 - hydraulic booster of the drive of the section of elevon; 33 - duplicated/backed up the hydraulic booster of the drive of the section of elevon; 33 - duplicated/backed up air steering jets along the bank on the left wing; 34 - duplicated/backed up air ducts to nozzles 33; 35 - air ducts of air bleed from the compressors of hoisting TPD; 36 - valve of jet-edge ailerons; 37 - duplicated/backed up air ducts to nozzles 11; III - arrangement/position of the assemblies of the mechanical feature of the system of lateral control in the front part of the fuselage; 38 - mechanism of the load of knob/stick during the lateral control; 39 - electrical mechanism of trimmer effect; 40 - control rod by the valve of jet-edge ailerons; 41 - electrohydraulic servodrive of control of the valve of jet-edge ailerons from the artificial stability installation; 42 - broken component/link; 43 - auxiliary hydraulic booster of lateral control; 44 - three-leg rocker of the connection of load system; 45 - outlet of the pull rod of roll control through the rotational axis of knob/stick during pitch control.

Key: (1). To the elevons. (2). Command/crew of transverse control. (3). To the hydraulic boosters of link of the sections of the elevons of left wing.

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It is necessary to focus attention on the fact that in systems, where possibility of service failure of part of hydraulic boosters of drive of one control surface (divided in section) is provided for, intake components/links of these hydraulic boosters must be connected up to general/common guide through "untying" mechanisms, which allow/assume possibility of moving of guide and intake components/links of operable hydraulic boosters to complete course of guide with wedging one or several hydraulic boosters in this control channel. These mechanisms are made in the form of the so-called spring rods with the sufficiently large preliminary tightening of spring or rocker with the limiting spring-cam clutch, designed for the transmission of the specific moment/torque (Fig. 19.46).

High-speed/high-velocity, and especially supersonic, are contemporary, aircraft frequently possess such adverse inherent characteristics of stability and controllability, that failure of automatic systems of improvement in these characteristics even with normally working hydraulic boosters can lead to impossibility of control. Therefore on such aircraft these automatic systems also repeatedly are reserved (they are noted for system with 3- and and 4-multiple redundancy).

Higher requirements according to reliability of system of control are imposed on aircraft of special types with air jet nozzle control (VTOL aircraft, VKS, etc.). That, for increasing the safety by the case of failure of one of the lift engines, from which air is selected/taken for the air jet nozzle control, the main lines of air bleed from each engine are united into the common collector/receptacle, the compressed air from which is fed/conducted to the valves, which control its issue through the nozzles (control jets). The main lines of the supply of air on the VTOL aircraft or steam-gas mixtures to VKS to the nozzles of jet guidance just as control valves, as a rule are duplicated/backed up (see Fig. 19.12 and 19.45). It is interesting to note that on the VTOL aircraft military designation/purpose the control valves with air exhaust through the nozzles are arranged/located nearer to the collector/receptacle. This somewhat decreases rapid action of this system of jet guidance in comparison with the system, where the valves are arranged/located directly in nozzles (compare with Fig. 19.9), but substantially raise life due to the decrease of the length of the air of main line, being under constant pressure.

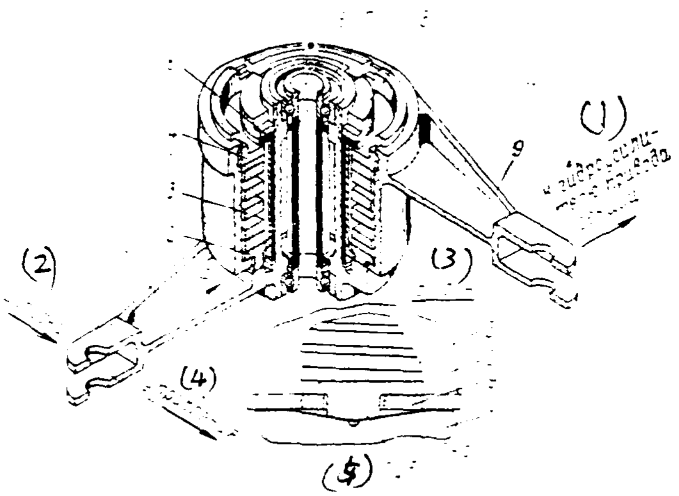

Fig. 19.46. Construction/design of "untying" rocker with limiting moment clutch: 1 - intake lever with axis/axle; 2 - check ring with teeth; 3 - spring; 4 - nut of control of tightening spring; 5 - bushing of slip; 6 - ball bearing; 7 - spacer; 8 - nut; 9 - output lever with sleeve/beaker.

Key: (1). To the hydraulic booster of the drive of section. (2). Guide. (3). Form. (4). Administrations. (5). Clamp.

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Failure of one of hoisting engines under conditions of vertical takeoff and landing presents large danger to VTOL aircraft. The angular position of aircraft in this case, as a rule must stabilize by the automatic systems (connection of the actuating mechanisms of these systems to the control line by VTOL aircraft shown, for example, in Fig. 19.9). The rapid compensation for vertical rod/thrust by means of thrust augmentation of remaining hoisting engines for unloading the pilot in the stressed situation must be accomplished/realized also automatically, for example, with the aid of the so-called group compensator of rod/thrust, which measures the pressure of exhaust gases of each of the lift engines of this group independent of pilot, moving control levers of rod/thrust in the direction of its increase during a pressure drop in exhaust nozzle of one of the engines (Fig. 19.47).

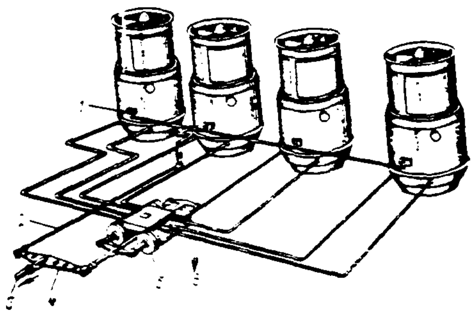

Fig. 19.47. Installation of automatic group compensator of rod/thrust of group of hoisting TRD of VTOL aircraft: 1 - throttle control lever of engine; 2 - control line to throttle levers; 3 - lever arm of control of vertical rod/thrust (in pilot); 4 - differential (summing) rocker; 5 - group compensator of rod/thrust; 6 - supply of pressure of exhaust gases TRD to computer of compensator.

* * *

In conclusion let us note that as a result of of increasing mechanization/lift-off device and automation of the systems of control of contemporary aircraft, increase in the required powers of hydraulic systems and redundancy for the purpose of increase safety the over-all payload ratio of control $\bar{G}_{упр}$ and of hydraulic systems $\bar{G}_{г.с}$ on the contemporary aircraft noticeably it increased in comparison with the aircraft with boosterless control.

Thus, processing statistical data for aircraft with boosterless control gives $\bar{G}_{упр}=0{,}006—0{,}015$; $\bar{G}_{г.с}=0{,}01—0{.}015$ (including control system by high-lift device of wing).

For aircraft with fully powered controls it is very difficult to conduct division between assemblies of system of control and hydraulic system. Therefore for these aircraft it is possible to accept $\bar{G}_{\text{упр-гс}} = 0.025-0.045$. As a result of the need of using the series/row of identical assemblies for the light and heavy aircraft larger values of over-all payload ratios, as a rule correspond to aircraft with the smaller takeoff weight.

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Appendices.

Appendix I.

STANDARD COMBINED WEIGHT OF AIRCRAFT.

I. Construction of the aircraft.

1. Wing (including hermetic sealing/pressurization of fuel compartments).
2. Fuselage (including hermetic sealing/pressurization of sections).
3. Tail assembly (including dorsal fins and washers).
4. Chassis/landing gear (including shutters/doors and fairings).
5. Weight of coloration.

II. Power plant.

1. Engines (fundamental and auxiliary, with systems of thrust reversal and mufflers).
2. Propellers with spinners and fastening (for aircraft with PD and TVD).
3. Means of engine installation:

a) engine nacelle with air ducts (if air duct is structurally connected with fuselage, then it relates to weight of fuselage);

b) pylons;

c) cowlings;

d) engine mounting and points of attachment;

e) mechanisms of rotation of engines (for VTOL aircraft).

4. Systems of engines:

a) starting system;

b) lubrication system;

c) control system of air intakes:

d) cooling system;

e) exhaust system; '

f) system of fire extinguishing;

g) throttle circuit;

h) system of control/check of work of engines;

i) de- icing system of power plant.

5. Fuel system:

a) fuel tanks with protectors;

b) adjusting fixtures of tanks;

c) propellant feed system (fuel lines and pumps);

d) pressurized system (inert gas);

e) system of automatic control of fuel consumption;

f) system of servicing;

g) system of emergency discharge;

h) system of servicing in flight.

III. Equipment and control.

1. Hydraulic system:

a) energy sources;

b) fixtures, tanks, communication;

c) working fluid;

d) fastening and operating devices/equipment.

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2. Pneumatic system:

a) energy sources;

b) fixtures, tanks/balloons, communication;

c) compressed air;

d) fastening and operating devices/equipment.

3. Electrical equipment:

a) generators and storage batteries/accumulators;

b) energy converters;

c) electric wiring, equipment, fastening and operating devices/equipment.

4. Control of aircraft:

a) control of controls and ailerons;

b) flap control, by slats, by interceptors/spoilers;

c) control of other mechanisms and aggregates/units (except power plant).

5. Radio equipment:

a) equipment for radio communication;

b) radio navigation equipment;

c) radar equipment;

d) system of automatic takeoff and landing.

Note. Radio equipment includes communications, antennas, equipment and devices/equipment of fastening.

6. Aeronautical equipment:

a) air navigation instruments;

b) control displays of work of systems and mechanisms (besides power plant);

c) autopilot;

d) comuter device/equipment, panels, instrument panels.

7. De-icing system.

8. System of heat and sound insulation of cabins/compartments and instrument compartments.

9. Survival facility and standard equipment:

a) seat of crew;

b) seat of passengers;

c) conditioning system;

d) oxygen system;

e) dress/lavatories;

f) kitchen and canteens (for passenger aircraft);

g) cloakrooms and baggage carriers (for passenger aircraft);

h) fire-fighting equipment of cabins/compartments;

i) ejection system of crew.

10. Special equipment, armament and armoring:

a) system of automatic control of work of equipment and construction of aircraft;

b) gun, machine guns with installation (without ammunition);

c) part of installation and equipment for rocket armament;

d) equipment bomber, fastening devices/equipment of active and passive protection;

e) armoring of crew and equipment;

f) photo-equipment, acquisition system and the like;

g) tiedown (fixed) equipment.

IV. Empty aircraft.

1. Construction/design of aircraft

2. Power plant.

3. Equipment and control.

V. Equipment official load.

1. Crew:

a) flight personnel with personal effect;

b) auxiliary composition (stewards, etc.).

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2. Water in canteen and wash-stands (passenger aircraft).

3. Unproduced fuel/propellant.

4. Oil.

5. Expendable in flight technical liquids (for example, anti-icing liquid).

6. Products in canteen of passenger aircraft (with containers).

7. Literature for passengers, carpets, covers for seats, first-aid kit, etc.

8. Rescue equipment (boats, emergency ladders, etc.).

9. Spare parts, on-board instrument, covers for assemblies of aircraft, signal flares.

10. Suspension tanks (without fuel/propellant), assemblies of removable suspensions for special loads, removable sanitary equipment.

11. Containers for luggage, loads and mail.

VI. Equipped aircraft.

1. Empty aircraft.

2. Equipment.

3. Official load.

VII. Fuel/propellant.

1. Expendable fuel/propellant.

2. Navigational reserve.

3. Fuel/propellant in supplementary and suspension tanks.

VIII. Payload.

1. Bombs, torpedo, rocket.

2. Ammunition (cartridges/ammunition, projectiles).

3. Special loads and jettisonable substances.

4. Equipment for photo-reconnaisance.

Key: (1a). Fighters, bombers are reconnaissance aircraft.

5. Landing force members with armament and parachutes.

6. Combat materiel, including pilot and main parachutes, platform.

Key: (5a). Military transport aircraft.

7. Passengers.

8. Luggage, loads, mail.

Key: (7a). Passenger aircraft.

9. Chemicals (agricultural aircraft).

IX. Full load.

1. Equipment official load.

2. Fuel/propellant.

3. Payload.

X. The takeoff weight of aircraft.

1. Empty weight.

2. Weight of full load.

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Appendix II.

APPROXIMATE ENUMERATION OF THE ELECTRONIC EQUIPMENT OF AIRCRAFT.

Heavy nonmaneuverable aircraft.

1. Communication equipment:

a) short-wave transceiver radio stations (two assemblies for duplication);

b) ultrashort-wave receiving-transmitting radio stations (two assemblies);

c) radiotelephone;

d) equipment for reception of official transfers.

2. Navigation equipment:

a) system of short-range navigation and automatic landing;

b) system of autonomous navigation (inertial, astronavigation);

c) radio compass;

d) radio altimeter;

e) marker receiver;

f) computer for optimization of flight conditions.

3. Radar equipment:

a) display system and obstacle detection (transport aircraft);

b) acquisition system and guidance;

c) control system by armament;

d) system of interferences/jammings and protection from attack.

4. Identification system (responder and interrogator).

5. Internal (onboard) radio communication:

a) loud-speaker device/equipment in cabins/compartments;

b) crew intercommunication equipment.

6. Onboard electronic equipment of automatic check of systems, mechanisms and construction of aircraft.

Maneuverable military aircraft.

1. Communication equipment:

a) ultrashort-wave and decimeter command of radio station;

b) short-wave communications radio station.

2. Navigation equipment:

a) system of short-range navigation and landing;

b) radio compass, radio altimeter, marker receiver.

3. Guidance system from ground-based control center (onboard system of reception of commands).

4. Radar equipment:

a) radar sight;

b) display system;

c) computer;

d) control system by weapon.

5. Identification system (responder and interrogator).

6. System of air signals and system "course vertical".

AD-A172 865 DESIGN OF AIRCRAFT (SELECTED CHAPTERS)(U) FOREIGN TECHNOLOGY DIV WRIGHT-PATTERSON AFB OH A A BADYAGIN ET AL. 22 SEP 86 FTD-ID(RS)T-0684-86 8/8

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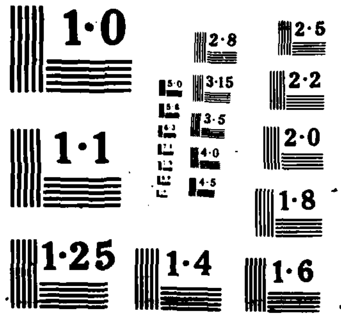
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Appendix III.

CHARACTERISTICS OF CONTEMPORARY AIRCRAFT ENGINES.

Characteristics of contemporary gas turbine engines (Table and Fig. III.1-III.10) Proposed below are intended for accomplishing necessary calculations in period of diploma design.

Altitude-speed characteristics TRD and TRDD.

Without being characteristics of any concrete/specific engines, data of characteristic, however, with degree of accuracy sufficient for their designation/purpose reflect altitude effect and flight speed on thrust and specific consumption of fuel of contemporary aircraft engines (on graphs they are given relative value of value P and c_p).

As it was shown by Chapter VII, characteristics TRD and TRDD have specific distinctive features, caused by the type of engine and by the maximum speed of flight, but with accomplishing of calculations to the diploma project these features in the altitude-speed characteristics can be disregarded/neglected.

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(1) Назначение¹	(2) Двигатели для дозвуковых пассажирских самолетов			(3) Двигатели для сверхзвуковых самолетов многоцелевого назначения			(4) Двигатели для СПС	
(5) Тип двигателя (6) Фирма	ТРДД (7) Пратт-Уитни	ТРДД Пратт-Уитни (7)	ТРДД Пратт-Уитни (7)	ТРДДФ (8) Роллс-Ройс	ТРД (9) Дженерал Электрик	ТРДДФ Пратт-Уитни (7)	ТРД (10) Бристоль Сидлей	ТРД (9) Дженерал Электрик
(11) Обозначение (марка)	JT8D (12) (США)	JT3D (12) (США)	JT9D (12) (США)	RB. 172/T260 (13) (Англия)	J79 (12) (США)	TF-30 (12) (США)	«Олимпус 593» (14) (Англия)	GE4 (США) (12)
(15) Степень двухконтурности	m = 1,1	m = 1,1	m = 5	m = 1,25		m = 1,3		
(16) Взлетный режим (с полным форсированием):								
(17) тяга в кгс				3150	[illegible]	9070	16 970	28 200
(18) удельный расход топлива в кгс/кгс·ч				1,75	1,85	2,0	1,68	1,72
(19) расход воздуха в кгс/с				50	76,5	113	190	280
(20) Взлетный режим (без форсирования):								
(17) тяга в кгс	6350	8165	19 750	2100	[illegible]	5200	15 900	[illegible]
(18) удельный расход топлива в кгс/кгс·ч	0,585	0,51	0,345	0,63	0,73	0,8	0,7	[illegible]
(19) расход воздуха в кгс/с	143	204	685	50	76,5	113	190	280
(21) Сухой вес в кгс	1405	1800	3575	580	1670	1590	2 800	1760
(22) Длина в м	3,7	3,98	7,05	3,1	4,6	5,5	3,6	7,82
(23) Диаметр (max) в м	1,08	1,35	2,43	0,81	1,1	1,2	1,4	1,88
(24) Самолет (стартовый вес, число двигателей)	(25) «Супер-Каравелла» (см. рис. III-1) (G_0 = 50 тс; $n_{дв}$ = 2), (34) Боинг-727 (G_0 = 77 тс, $n_{дв}$ = 3), (35) Дуглас DC-9 (G_0 = 41 тс, $n_{дв}$ = 2), (36) Боинг-737 (G_0 = 45 тс; $n_{дв}$ = 2)	(26) Боинг-707 (G_0 = 141 тс; $n_{дв}$ = 4), (37) Дуглас DC-8 (G_0 = 140,5 тс, $n_{дв}$ = 4)	(27) Боинг-747 (см. рис. III-2) (38) (G_0 = 322 тс, $n_{дв}$ = 4)	(28) «Ягуар» (G_0 = 10 тс, $n_{дв}$ = 2)	F-104 (31) (G_0 = 10,2 тс, $n_{дв}$ = 1), (32) «Фантом»-II (G_0 = 21 тс; $n_{дв}$ = 2)	F-111 (G_0 = 31,8 тс; $n_{дв}$ = 2), (33) «Корсар»-II (см. рис. III-3) (G_0 = 11,8 тс; $n_{дв}$ = 1)	(29) «Конкорд» (G_0 = 170 тс, $n_{дв}$ = 4)	(30) Боинг-2707 (G_0 = 290 тс, $n_{дв}$ = 4)

Key: (1). Designation/purpose ¹.

FOOTNOTE ¹. This table of the fundamental characteristics of most modern foreign TRD and TRDD is comprised based on materials, published in the open foreign literature. ENDFOOTNOTE.

(2). Engines for subsonic passenger aircraft. (3). Engines for supersonic aircraft of multipurpose designation/purpose. (4). Engines for SPS. (5). Type of engine. (6). Firm. (7). Pratt-Whitney. (8). Rolls-Royce. (9). General Electric. (10). Bristol-Siddeley. (11). Designation (brand). (12). (USA). (13). (England). (14). (England). (15). Bypass ratio. (16). Takeoff conditions (with complete boosting). (17). thrust in kg. (18). specific expenditure/consumption of fuel in kg/kg·h. (19). air flow rate in kgf/s. (20). Takeoff conditions (without boosting). (21). empty weight in kg. (22). Length m. (23). Diameter (max) m. (24). Aircraft (launching weight, number of engines). (25). "Super-Caravelle" (see Fig. III-1) (G_0=50 t; .... (26). Boeing-707 (G_0=141 t; .... (27). Boeing-747 (see Fig. III-2) (G_0=322 t, ... (28). "Jaguar" (G_0=10 t, ... (29). "Concorde" (G_0=170 t; .... (30). Boing- 2707 (G_0=270 t; .... (31). Douglas DC-8 (G_0=140.5 t; .... (31a). t. (32). "Phantom" -II (G_0=21 t; .... (33). "Corsair" -II (see Fig. III.3) (G_0=14.8 t; .... (34). Boeing-727 (G_0=77 t; .... (35). Douglas DC-9 (G_0=41 t; .... (36). Boeing-737 (G_0=45 t; ...).

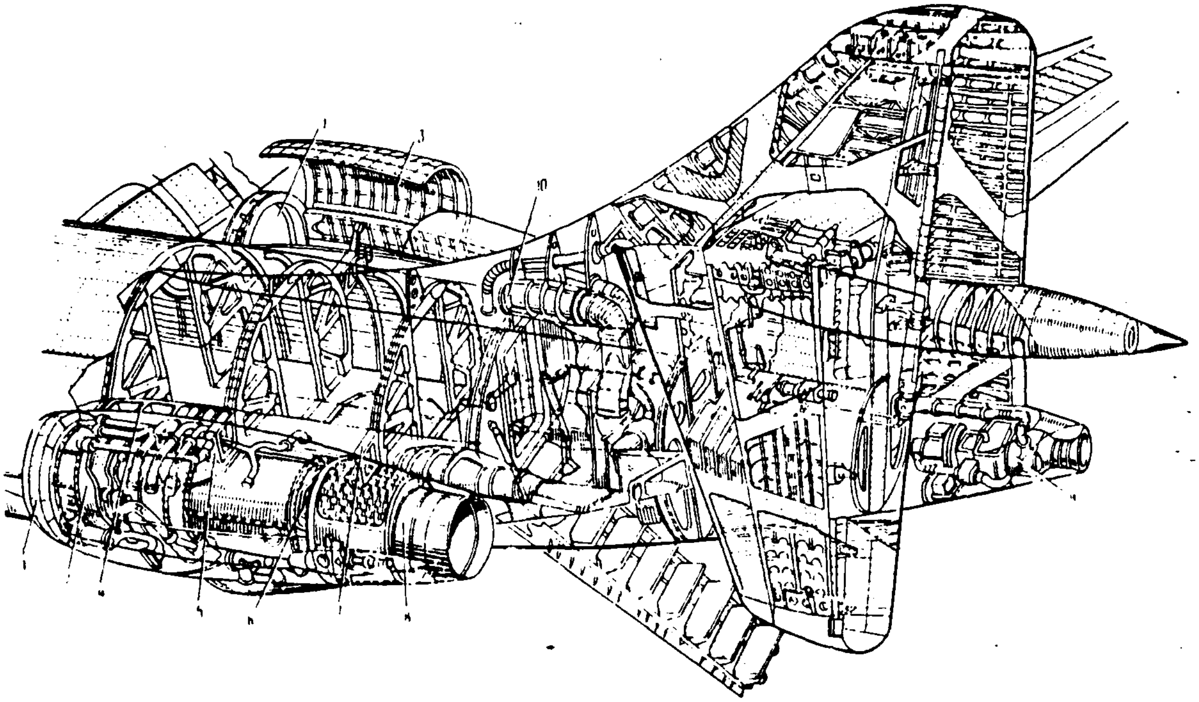

Fig. III.1. Layout TRDD Pratt-Whitney JT8D on the fuselage of French aircraft "Super Caravelle": 1 - air intake with anti-icing system; 2 - engine; 3 - cowling; 4 - strut of fastening engine; 5 - Main point of attachment of engine; 6 - reverser of thrust; 7 - rear point of attachment of engine; 8 - nozzle; 9 - VSU; 10 - air intake VSU.

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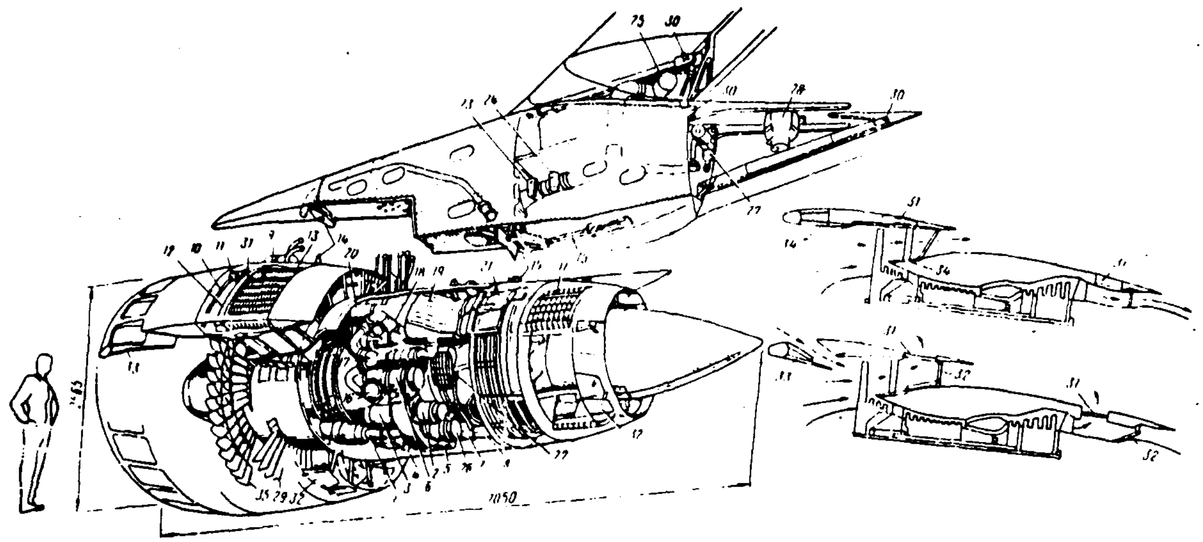

Fig. III.2. Layout TRDD of Pratt-uitin JT9D and the schematic of the work of the reversers of thrust and intake shutters/doors (aircraft of Boeing-747): 1 - drive shaft of auxiliary units; 2 - coaxial constant-speed drive; 3 - generator with a power of 60 kV•A, 4 - pump and the control unit; 5 - centrifuge; 6 - oil tank; 7 - drive of the reverser of the thrust of gas generator; 8 - cable run to the reverser; 9 - compressed air motor of the drive of the reverser of the thrust of fan; 10 - flexible drive shaft; 11 - gearbox and jack; 12 - moving ring; 13 - roller slide; 14 - unit of engine mount; 15 - component/link, which receives thrust; 16 - air bleeding; 17 - check valve of selection from 8th step/stage; 18 - cutoff the valve of selection from the 15th step/stage; 19 - primary heat exchanger; 20 - air from the fan to the heat exchanger, 21 - bleeder of air from the fan; 22 - valve of depressurizing; 23 - block of the temperature control of passed air; 24 - cutoff and control valve; 25 - main line of pneumatic system; 26 - pneumo-starter; 27 - given by compressed air motor hydraulic pump; 28 - hydraulic tank; 29 - stators; 30 - bolts; 31 - grid/cascade of reverser 32 - the shutter/door of reverser; 33 - intake shutters/doors; 34 - acoustic panels; 35 - fan; 36 - stators.

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Therefore the given below (see Fig. III-6) characteristics of supersonic GTD (designed on $M_{max} \approx 3.5$) can be in this case considered as characteristics TRD (and TRDF), and also TRDD with the small bypass ratio (without afterburning and afterburner in the secondary circuit), intended for the contemporary multipurpose aircraft and the aircraft with cruising speed, which corresponds to $M_{крейс} \approx 2{,}0—3{,}0$. The altitude-speed characteristics of lift engines for VTOL aircraft and STOL can be identified with the characteristics of subsonic TRDD (Fig. III-4).

Altitude-speed characteristics of subsonic TRDD with high bypass ratio (m=5-8) are given separately (Fig. III-5).

Since on graphs are given relative value of thrust and specific consumption of fuel (referred to starting values of indicated values), consequently, dependences P=f(M, H) and $c_p=f(M, H)$ can be obtained for any selected value P_0 and c_{p0}.

As is known, at heights/altitudes, large 11 km (during constant/invariable regime and M=const); specific fuel consumption remains constant, and engine thrust varies in proportion to to density or pressure of atmospheric air; therefore it is determined by simple conversion. For example, for any value of Mach number:

$$P_{13}=P_{11}\frac{p_{13}}{p_{11}} \quad \text{or} \quad P_{20}=P_{15}\frac{p_{20}}{p_{15}}; \quad c_{p11}=c_{p13}=c_{p15}=c_{p20}.$$

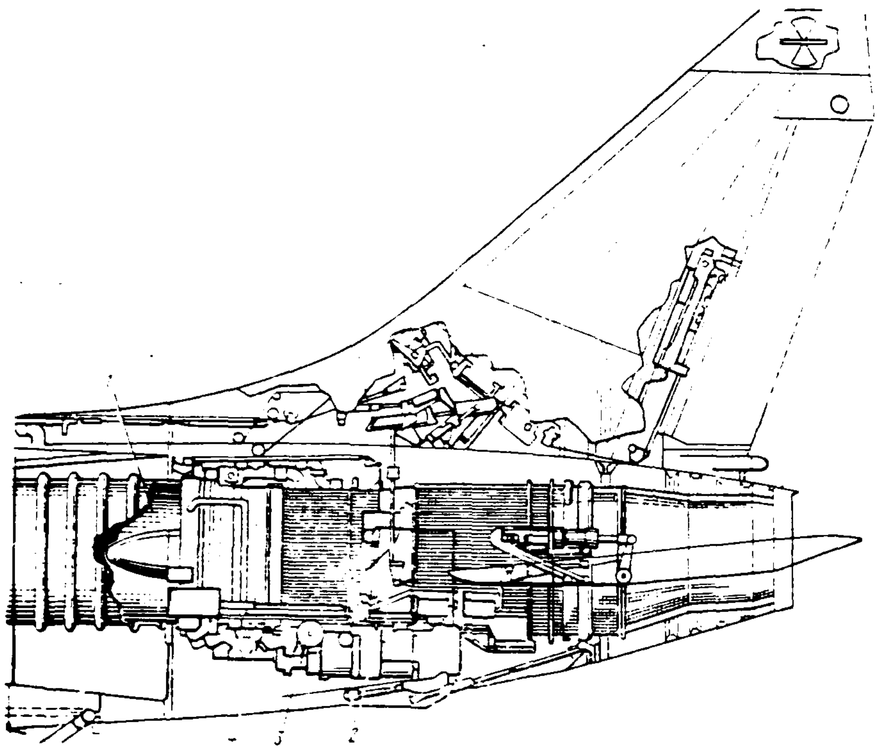

Fig. III.3. Layout TRDD Pratt-Whitney TF-30 on the fighter-bomber "corsair" of II (USA): 1 - air duct; 2 - engine; 3 - starter; 4 - generator.

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Transition/transfer from climb regime to regime of cruise (value $c_{p\,\text{крейс}}$ they are given below) is accomplished/realized by throttling/choking engine (reduction in revolutions). In the cruise number of revolutions $n=(0{,}8-0{,}9)n_{\max}$.

For supersonic cruise at high altitudes is used commonly regime of partial afterburning (as a rule low afterburning). Number of revolutions in this case also somewhat is reduced.

They are given below:

- dependence of specific consumption of fuel (kerosene) of aviation VRD on cruising Mach number (Fig. III.7). for the cryogenic fuel/propellant (liquid hydrogen) the indicated dependence will be the same, only

$$c_{p(вод)} \approx \frac{c_{p(кер)}}{2,5};$$

- the dependence of the starting flow rate per second of air TRD and TRDD on the maximum boost for launching (Fig. III-8). A sharp increase of the flow rate of air of large TRDD is explained by the high bypass ratio of these engines (m=5-8);

- dependence of the given flow rate of air TRD and TRDD on the total stagnation temperature (Fig. III-9);

- dependence of diameter TRD and TRDD on the boost for launching (Fig. III.10). A sharp increase of the diameter of large TRDD is explained by the high bypass ratio of these engines (m=5-8).

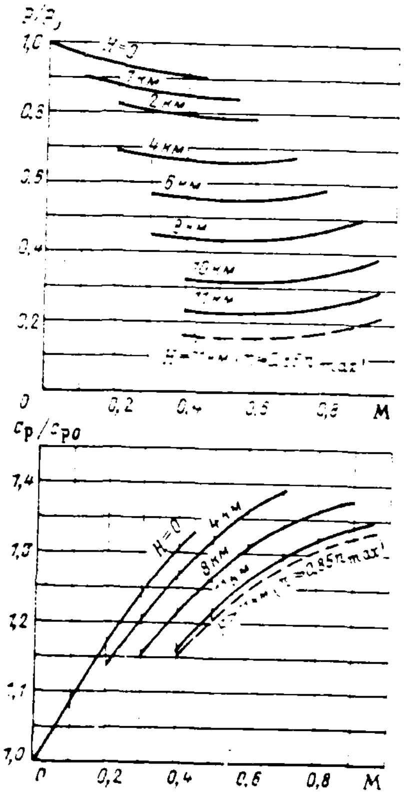

Fig. III.4.

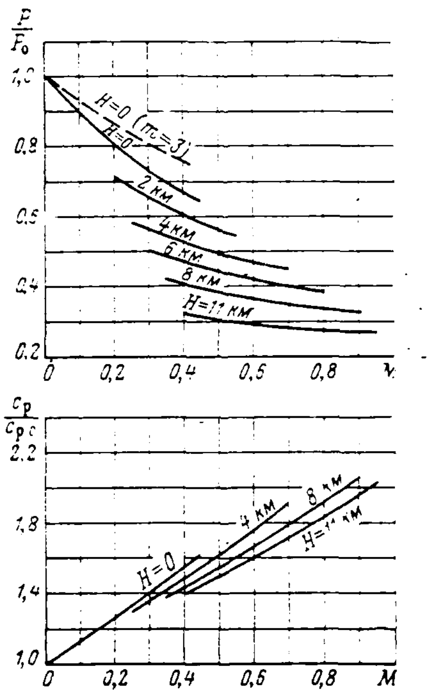

Fig. III.5.

Fig. III.4. Altitude-speed characteristics TRDD (m=0-2), designed for the subsonic cruise.

Fig. III.5. Altitude-speed characteristics TRDD (m=6), designed for the subsonic cruise.

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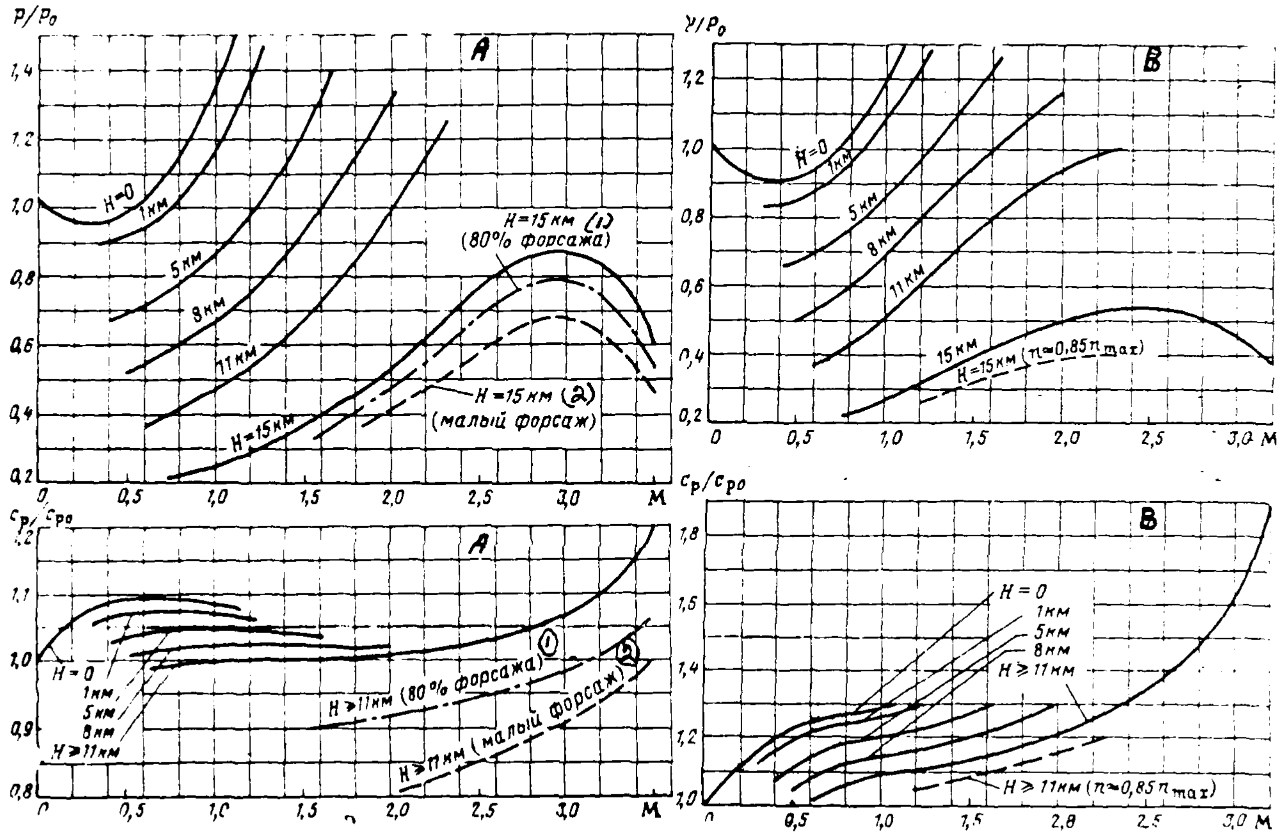

Fig. III.6. Altitude-speed characteristics GTD, designed for the supersonic cruise and the maximum speed, which corresponds to number M=3-3.5: A - afterburner; B - maximum rating without afterburning. Key: (1). Afterburning. (2). low afterburning.

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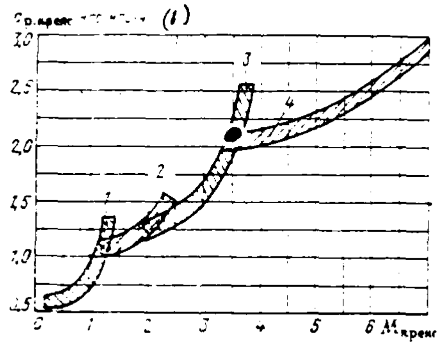

Fig. III.7.

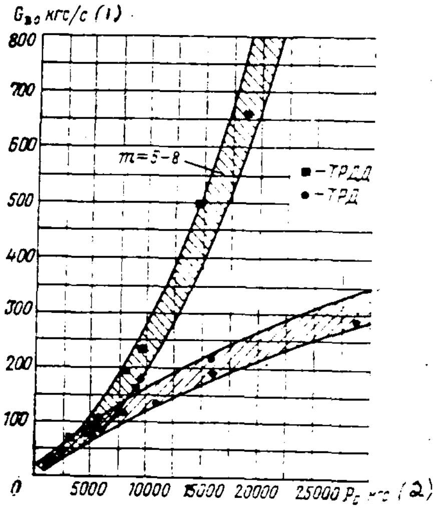

Fig. III.8.

Fig. III.7. The dependence of the specific consumption of fuel (kerosene) of aviation VRD on cruising Mach number: 1 - TRDD $(m=1\text{–}8;\ \pi_к=20\text{–}27)$; 2 - TRD and TRDD $(m=0{,}5\text{–}1{,}5;\ \pi_к=9\text{–}10)$; 3 - TRD and TRDD $(m=0{,}5\text{–}1{,}5;\ \pi_к=3\text{–}4)$; 4 - PVRD with the subsonic combustion.

Key: (1). kg/kg·h.

Fig. III.8. Dependence of the starting second flow rate of air TRD and TRDD on maximum boost for launching тяги $\left(\bar{G}_{в.пр}=\frac{G_{в.пр}}{G_{в0}}\right)$

Key: (1). kgf/s. (2). kg.

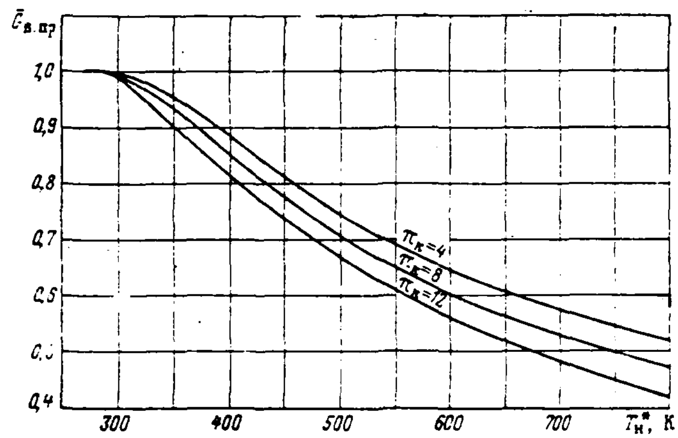

Fig. III.9. Dependence of the given air flow rate for different TRD and TRDD on temperature of total stagnation ($G_{в0}$ - starting flow rate per second of air).

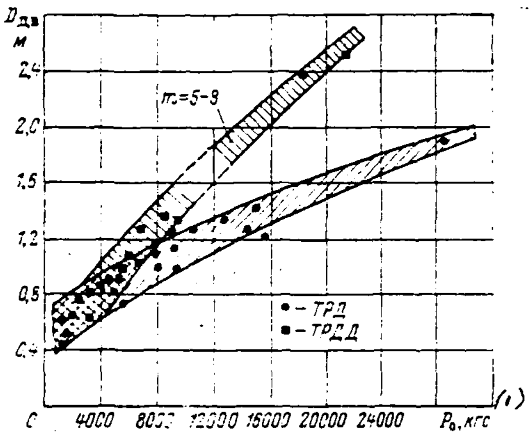

Fig. III.10. Dependence of diameter TRD and TRDD on the boost for launching.

Key: (1). kg.

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Appendix IV.

FUNDAMENTAL PARAMETERS AND CHARACTERISTICS OF AIRCRAFT.

Are given below basic datum and general views in three projections of multipurpose aircraft, tactical aircraft, military transport, experimental and passenger (Fig. IV-1-IV.21). These data can be useful during the diploma design.

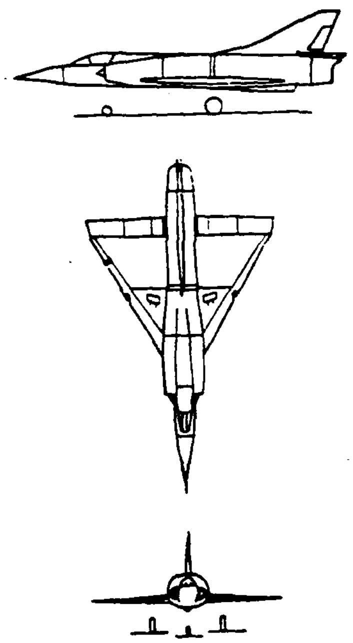

Fig. IV.1.

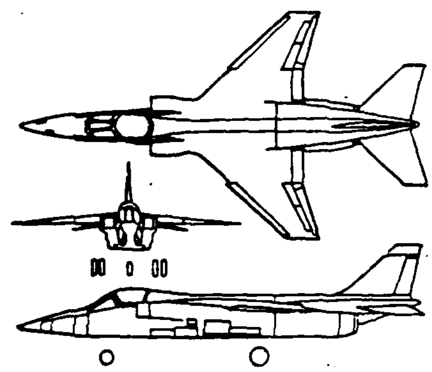

Fig. IV.2.

Fig. IV.1. Fighter "Mirage" IIIC (France).

Fig. IV.2. Fighter- bomber "Jaguar" (England-France).

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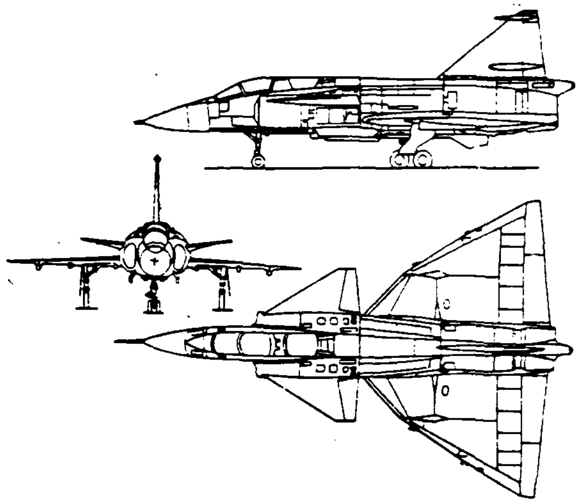

Fig. IV.3. Multipurpose fighter SAAB-37 "Viggen" (Sweden).

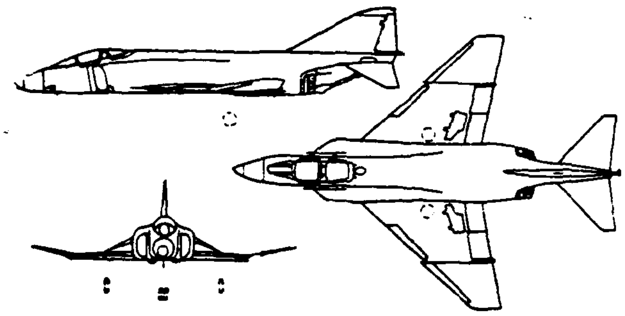

Fig. IV.4. Multipurpose fighter F4H-1 "Phantom" II (USA).

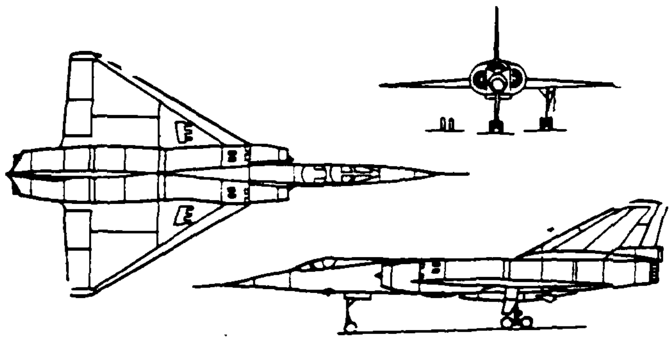

Fig. IV.5. The supersonic bomber "Mirage" IVA (France).

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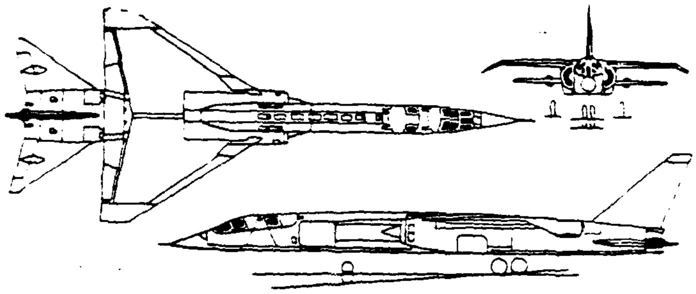

Fig. IV.6. Multipurpose aircraft TSR-2 (England).

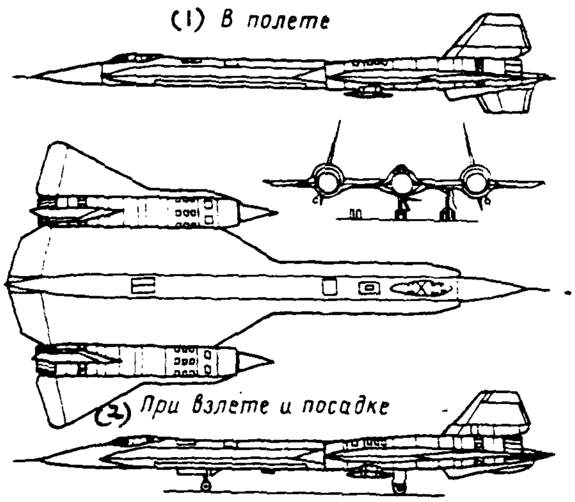

Fig. IV.7. Fighter-interceptor Lockheed YF-12A (USA).

Key: (1). In flight. (2). During take off and landing.

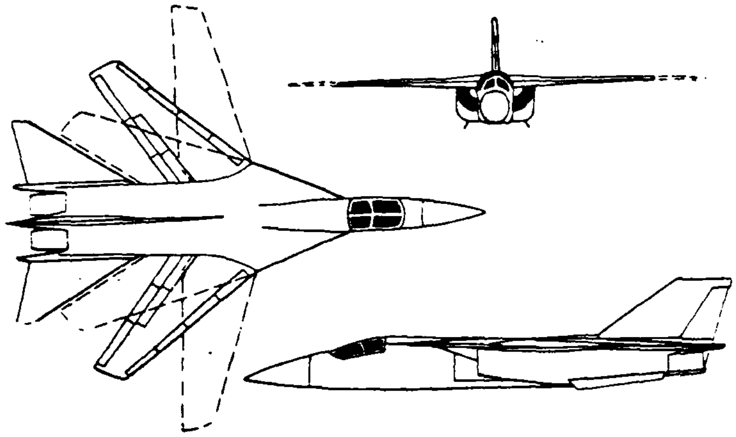

Fig. IV.8. The multipurpose aircraft of General-Dynamics F111A (USA).

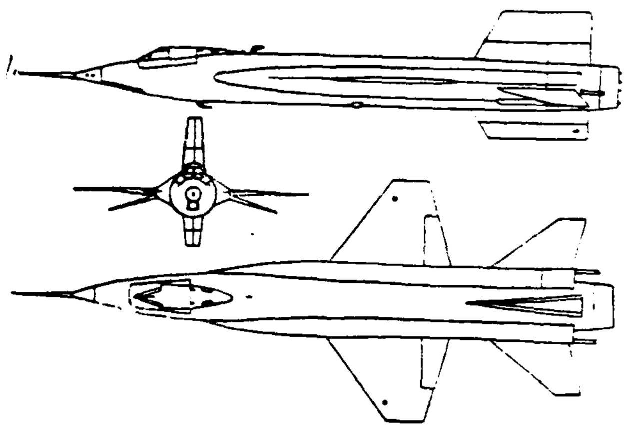

Fig. IV.9. Experimental aircraft of North American X-15 (USA).

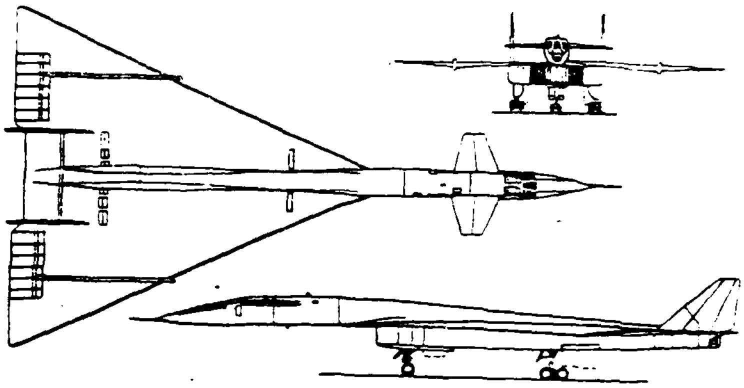

Fig. IV.10. Experimental aircraft of North American XB-70A (USA).

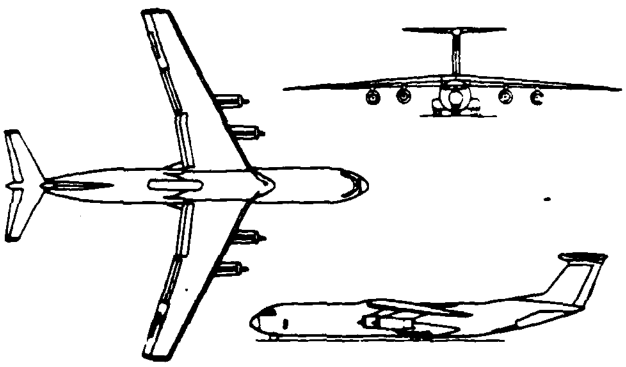

Fig. IV.11. Military transport aircraft Lockheed C- 141A (USA).

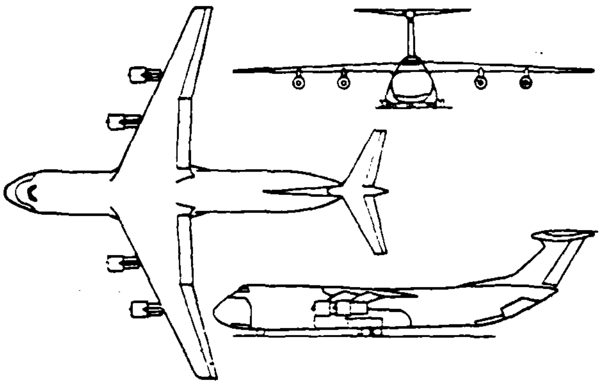

Fig. IV.12.

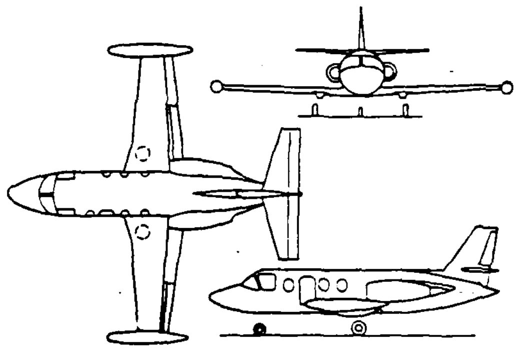

Fig. IV.13.

Fig. IV.12. Military transport aircraft Lockheed C-54 (USA).

Fig. IV.13. Administrative and passenger aircraft Piaggio-Douglas PD-808 (Italy - USA).

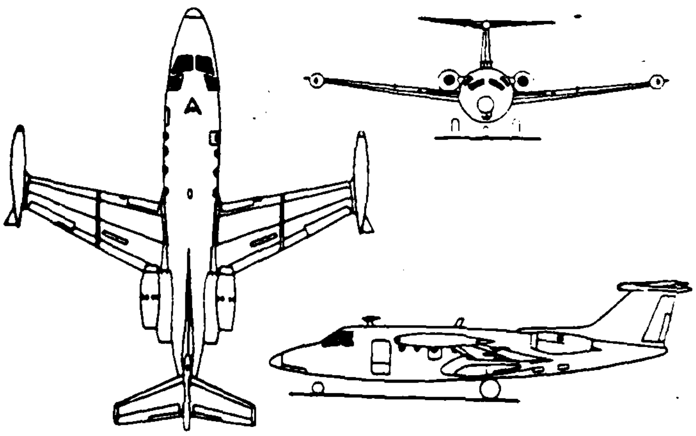

Fig. IV.14. Administrative and passenger aircraft HFB-320 of "Hansa" (FRG).

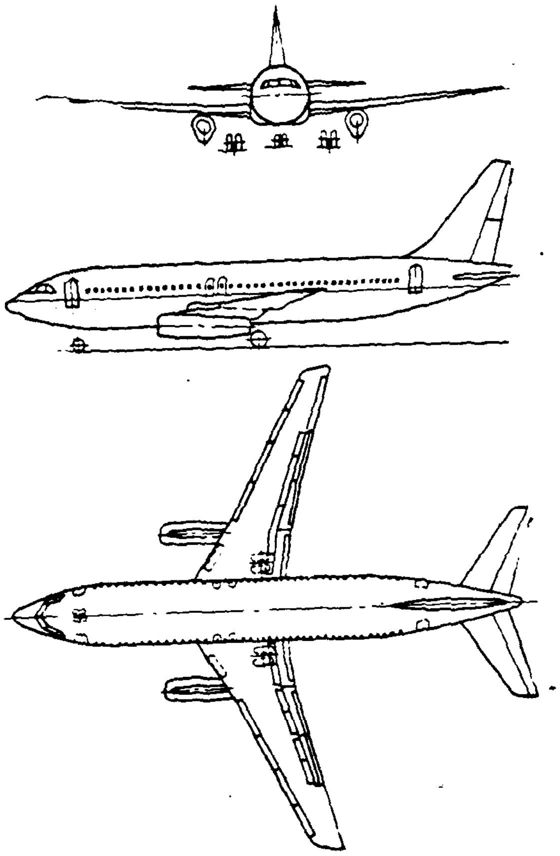

Fig. IV.15. Passenger aircraft for the air lines of the low extent of Dassault "Mercury" (France).

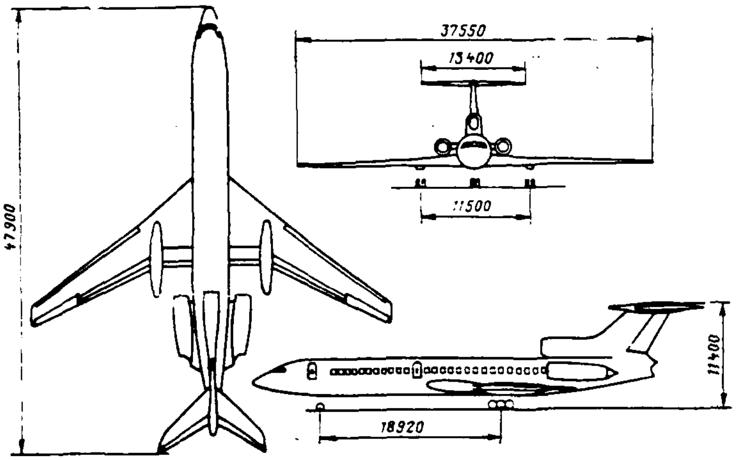

Fig. IV.16. Passenger aircraft for the air lines of the average/mean extent of Tu-154 (USSR).

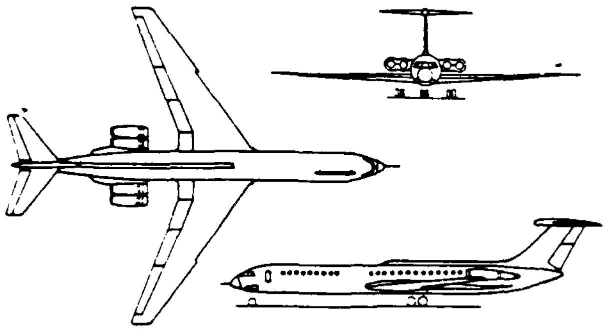

Fig. IV.17. Passenger aircraft for the air lines of the large extent of Il-62 (USSR).

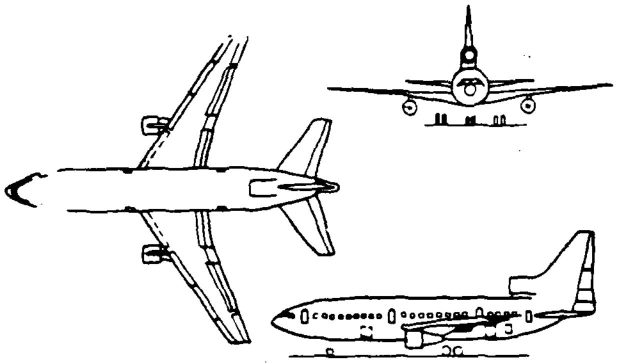

Fig. IV.18. Aircraft of high passenger capacity Lockheed L-1011 (USA).

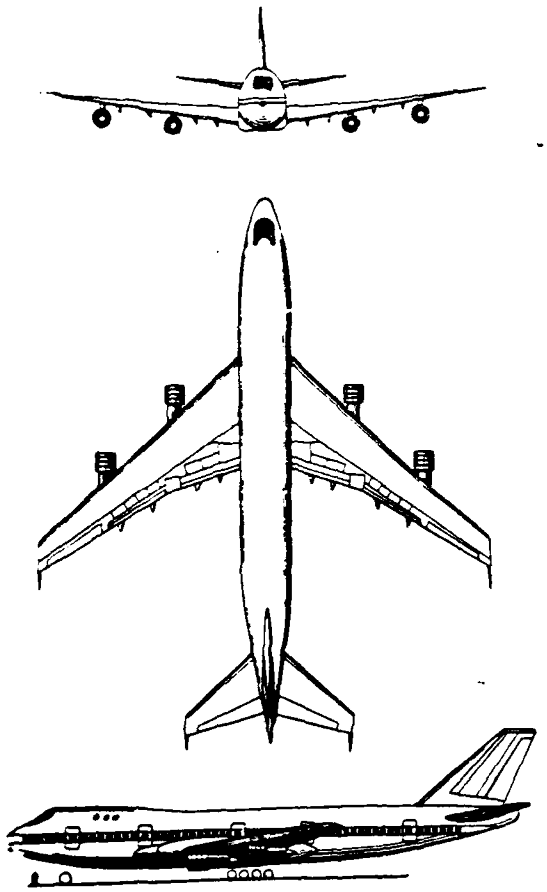

Fig. IV.19. Aircraft of high passenger capacity Boeing-747 (USA).

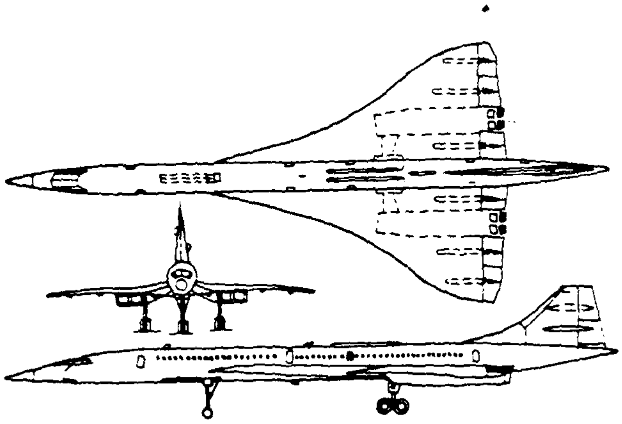

Fig. IV.20. Supersonic passenger aircraft "Concorde" (France-England).

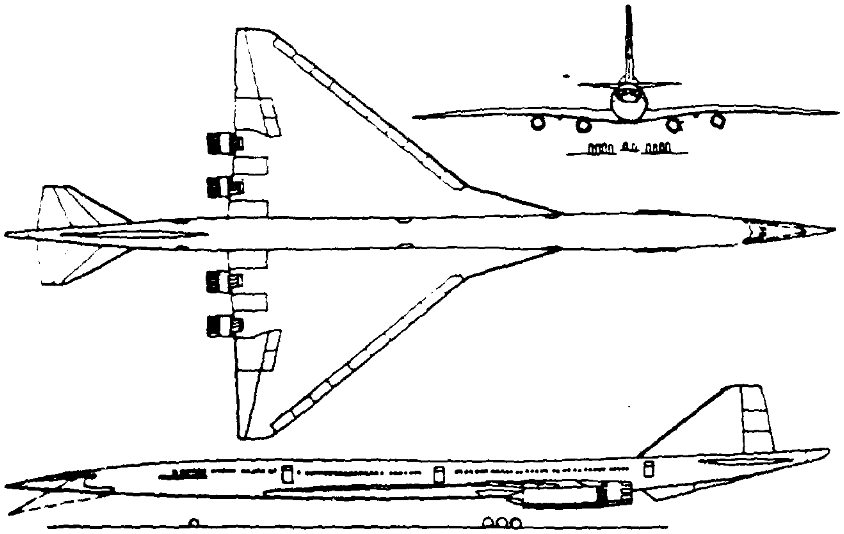

Fig. IV.21. Supersonic passenger aircraft the Boeing-2707-300 (project, the USA).

Рис. 506.

Multipurpose Aircraft (fighters and bombers)

Параметры и характеристики (1)	«Мираж» III C (Франция) (2)	«Ягуар» (Англия—Франция) (3)	«Виггень» (Швеция) (4)	«Фантом»-II (США) (5)	F-106A Дельта Дарт (США) (6)	«Виджилент» A3J-1 (США) (7)
(8) Взлетный вес в кгс	8950	9980	16 000	24 800	16 000	27 000
(9) Тяга двигателей на старте с форсажем (без форсажа)	7500 (4250)	6300 (4200)	12 000 (7 000)	15 000 (9 500)	11 000 (7 800)	15 400 (9 900)
(10) Число двигателей	1 дв. ТРДФ + ЖРД (11)	2 дв. ТРДД (12)	1 дв. ТРДД	2 дв. ТРДФ (13)	1 дв. ТРДФ (13)	2 дв. ДРДФ (14)
(15) Тяговооруженность с форсажем (без форсажа)	0,84 (0,475)	0,632 (0,424)	(12) 0,75 (0,44)	0,604 (0,383)	0,69 (0,49)	0,57 (0,367)
(16) Площадь крыла в м²	35	24,2	35	49,2 (есть УПС) (17)	61,5	65 (есть УПС) (17)
(18) Нагрузка при взлете в кгс/м²	256	412	457	505	260	415
(19) Размах в м	8,2	8,5	10,61	11,7	11,62	16,15
(20) Длина в м	13,3	15,5	16,31	17,76	21,56	22,25
(21) Стреловидность крыла по передней кромке	60°	45°	42°—58°	50°	60°	45°
(22) Относительная толщина крыла у корня (на конце) в %	4,5 (3,5)			5,1 (22) (средняя)	4,5	3,5
(23) Вес пустого в кгс (%)	6350 (71)			12 700 (51,2)	11 800 (73,7)	12 300
(24) Вес топлива во внутренних баках в кгс (%)	1700 (19)			6 200 (25)		
(25) Максимально допустимый посадочный вес в кгс	6600			17 200		
(26) Максимальная скорость в км/ч						
$H = 0$	1100	M = 1,1	M = 1,0			
$H \geq 11$ км	2350 (M = 2,2)	M = 1,7	M = 2,0	M = 2,4	M = 2,1	M = 2,1
(27) Практический потолок:				21,6	>17	21
(28) — без ЖРД в км	18					
(29) — с ЖРД в км	22					
(30) Дальность полета (топливо только во внутренних баках) в км	1300	1250		>2 400	2 300	4 000
(31) Дальность перегоночная в км	3600	4500		3 700	4 350	>6 000
(32) Взлетная дистанция в м	900	715		2 000	1 300	
(33) Посадочная дистанция в м	950	850		1 500		
(34) Вооружение (П — пушки; УРС — управляемые реактивные снаряды; РС — реактивные снаряды; Б — бомбы)	П = 2×30; 2×УРС; РС = 72 шт. (35) Б = 1350 кгс (36)	УРС; РС	УРС; Б	УРС (36) Б = 6000 кгс	УРС	УРС; РС; Б

Key: (1). parameters and characteristic. (2). "Mirage" III C (France). (3). "Jaguar" (England-France). (4). "Viggen" (Sweden). (5). "Phantom" - II (USA). (6). Delta Dart. (7). Vigilant. (8). Takeoff weight in kg. (9). Engine thrust at the start afterburner (without afterburning). (10). Number of engines. (11). eng. TRDF +JRD. (12). eng. TRDD. (13). eng. TRDF. (14). eng. DRDF. (15). Thrust-weight ratio afterburner (without afterburning). (16). Wing area in m². (17). there are UPS. (18). Load during the takeoff in kgf/m². (19). Spread/scope m. (20). Length m. (21). Sweepback of wing on leading edge. (22). Wing chord ratio at the root (at the end) in %. (23). Weight of empty in kg (%). (24). Fuel load in the internal tanks in kg (%). (25). Maximum permissible landing weight in kg. (26). The maximum speed in km/h. (27). The service ceiling.

(28). without ZhRD in km. (29). s ZhRD in km. (30). Flying range (fuel/propellant only in the internal tanks) in km. (31). Distance is distillation in km. (32). Takeoff distance m. (33). Landing distance m. (34). Armament (P - gun; URS - guided missiles; RS - rocket projectiles; B - bomb). (35). pcs. (36). kg.

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Tactical aircraft, experimental and military transport.

Параметры и характеристики (1)	(2) Тактические самолеты				(5) Экспериментальные самолеты		(6) Военно-транспортные самолеты	
	«Мираж» IVA (Франция) (3)	TSR-2 (Англия) (4)	YF-12A (США)	F-111A (США)	X-15 (США)	XB-70A (США)	C-141A (США)	C-5A (США)
Взлетный вес в тс (7)	30	40	45	31,8	15,14	250	143,0	323
Тяга на старте (8)								
$P_{оф}$ в тс (9)	13,6	30	29	18		81,5		
$P_{о}$ в тс (9)	9,8	18,2			26,1	54,2	38,1	74,4
Число двигателей (10)	2×ТРДФ	2×ТРДФ	2×ТРДД	2×ТРДД	1×ЖРД	6×ТДРФ	4×ТРДД	4×ТРДД
$\bar{P}_{оф}$	0,454	0,750	0,615	0,565		0,326	—	—
$\bar{P}_{о}$	0,327	0,455			1,72		0,265	0,23
S в м²	63	60,5	120	48,8—58,74	18,6	585	300	576
$p_{о}$ в кгс/м² (12)	477	660	375	542—652	815	427	479	560
Размах в м (13)	11,8	10,7	15	19,2—9,74	6,7	32	48,8	67,8
Длина в м (14)	23,5	27,4	28	22,4	15,24	56,4	44,2	75,0
Удлинение крыла λ (15)	2,21	1,91	1,90	7,56—1,34	2,42	1,75	7,94	7,98
χ							25°	25°
$\chi°_{п.к}$	60	60	60	16—72,5	35	65		
$\bar{c}_{о}$ в % (11)	4,5	~4		3,5—12	5	~3 (16)	13	12
$\bar{c}_{к}$ в % (11)	3,5	(средняя) (16)				(средняя)	10	11
$G_{пуст}$ в тс (9)	14,0			17,87	6,07	107,5	61,2	144,5
$\bar{G}_{пуст}$ в % (11)	46,6			56,2	40	43	42,6	44,7
$G_{т}$ в тс (9)	11,5			~6	8,33	136	48	76
$\bar{G}_{т}$ в % (11)	38,4			18,9	55,0	55	33,5	23,5
(17) Максимальный $G_{п.н}$ в тс	3				0,68	10	32	99,8
$\bar{G}_{п.н}$ в % (11)	10				4,5	4	22,3	30,9
(18) Максимальный					6,6		117	288
$G_{взл.доп}$ в тс (9)							—	
(19) V (или M) в км/ч, $H=0$ км	1220	M = 1,5		M = 1,1				
$H \geq 11$ км	M = 2,2	M = 2,2	M = 3	M = 2,5	M = 6—7	M = 3	815	815
$H_{практ}$ в км		18,3	~30	~18	>100	20—25	12,2	
$L_{расч}$ в км	~3000	3200		~3200		12 000	6 700	5 600
$L_{макс}$ в км		6400		6400		14 500	10 000	10 200
$L_{разб}$ в м	~1000	550		500			1 200	
$L_{проб}$ в м	~700	600		600	1400	2 000	535	
$L_{взл.дист}$ в м				900			1 600	2 285
$L_{пос.дист}$ в м							1 130	1 220
(20) $V_{пос}$ в км/ч				200	370			
(21) Вооружение	Б; РС	УРС; РС; Б		П; УРС; Б	—		—	
(22) Примечания		УПС на закрылках (23)					$L_{ВПП}$ = 1800 м	

Key: (1). parameters and characteristic. (2). Tactical aircraft. (3). "illusion" IVA (France). (4). England. (5). Experimental aircraft. (6). Military transport aircraft. (7). Takeoff weight in t. (8). Thrust at the start. (9). V t. (10). Number of engines. (11). V. (12). V kgf/m². (13). Spread/scope m. (14). Length m. (15). Wing aspect ratio. (16). Average/mean. (17). Maximum ... in t. (18). Maximum. (19). V (or M) in km/h, H=0 km. (20). V km/h. (21). Armament. (22). Notes. (23). UPS on the flaps.

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Passenger jet aircraft.

Параметры и характеристики (1)	PD-808 (Италия — США) (2)	HFB-320 «Ганза» (ФРГ) (3)	Дассо «Меркурий» (Франция) (4)	Ту-154 (СССР) (5)	Ил-62 (СССР) (6)	Локхид L-1011 (США) (7)	Боинг-747В (США) (8)	«Конкорд» (Франция — Англия) (9)	Боинг-2707-300 (проект, США) (10)
G_0 в тс (11)	7,5	8,2	52,5	84	154,5	186	352	148	340,2
P_0 в тс (11)	2,38	2,58	14	29,5	42	53,8	86,5	63,6	121,5
$n_{дв}$	2×ТРД	2×ТРДД	2×ТРДД	3×ТРДД	4×ТРДД	3×ТРДД	4×ТРДД	4×ТРДФ	4×ТРДФ
Двухконтурность ТРДД (12)	0	0	1,05			5,0	5,0		
$\bar{P}_0$	0,317	0,315	0,268	0,339	0,272	0,29	0,245	0,43	0,357
S в м² (13)	20,9	30,0	116	180	282	321	511	357	730
p_0 в кгс/м² (14)	359	272	452	467	548	575	630	415	465
Размах в м (15)	11,42	14,42	30,55	37,55	43,3	47,0	59,8	-	43,2
Длина в м (16)	12,0	16,61	34,0	47,9	53,12	53,34	70,5	58,3	85,34
λ	6,25	6,90	8,06	7,0	6,63	6,88	7,0		2,55
$\chi°$	0	—15	25	35	35	35	37,5	$\chi_{пк}$ = 65° (средний)	$\chi_{пк}$ = 50,5°
$\bar{c}_0$	0,09	0,13	0,127	0,120			0,124	(17) 0,0297	0,031
$\bar{c}_a$	0,09	0,10	0,10	0,082			0,08	0,0215	(средняя) (18)
$D_ф$ в м (13)	1,8	2,06	3,9	3,8	3,93 (средний)	5,97	6,6	3,00 (средний)	~4
$\lambda_ф$	6,66	7,55	8,46	11,4	(17) 13	8,7	10,6	(17) 19,4	21,4
$n_{пас}$	7	12	155	158	186	345	490	136	321
Шаг кресел в см (19)			76	75	75		86,5	86	
(20) Высота пассажирской кабины в м	1,5	1,55	2,19			2,59	2,54	1,96	
(21) Длина пассажирской кабины в м	3,75	4,6	20,51			41	57,9	30	53,8
$G_{пуст}$ в тс (11)	4,2	4,42	25,45	44,0		~95	~147	~70	131
$\bar{G}_{пуст}$ в % (13)	56	53,8	48,5	52,4		51,2	45,7	47,3	38,5
$G_{т.расч}$ в тс (13)	2,25	2,38	7,0				105		184,6
$\bar{G}_{т.расч}$ в % (13)	30	29	14,2				32,5		54,2
(22) Максимальный $G_{ком}$ в тс	0,880	1,2	16	16,0	23,0	40	65,4	13	22,2
(23) Максимальный $\bar{G}_{ком}$ в %	11,8	14,6	30,5	19,1	14,9	21,5	20,3	8,8	6,53
(24) Максимальный $G_{пос}$ в тс	7,5	7,62	47,0	-		158	256		195
$V_{крейс}/H$ в $\frac{км/ч}{км}$ (25)	650/10	700/6	830/6	900/10	850/10	960/9	935/11	M-2,1/18—20	M — 2,7/18—21
$L_{расч}$ в км / $G_{ком}$ в тс (26)	2000/0,88	2000/1,2	900/15,2	2900/16			6500/65,4	6200/12	6680/22,2
L_{max} в км / $G_{ком}$ в тс (26)	3200	3200	1715/10,9	5600/5,8	9200	5300/17	9250/34,5		
$V_{зах}$ в км/ч (27)	206	180	232	230		240	240		270
$L_{разб}$ в м (13)	760	860	800	900	1800				
$L_{проб}$ в м (13)	590	435		700	1000				
$L_{впп}$ в м (13)	1450	1340	2180	1450		3000	3400	3000	3290

Key: (1). parameters and characteristic. (2). Italy-USA. (3). "Hansa" (FRG). (4). Lasso "Mercury" (France). (5). Tu-154 (USSR). (6). Il-62 (USSR). (7). Lockheed L-1011 (USA). (8). Boeing-747B (USA). (9). "Concorde" (France-England). (10). Boeing-2707-300 (project, USA). (11). V t. (12). Bypass configuration TRDD. (13). V. (14). V kgf/m². (15). Spread/scope m. (16). Length m. (17). Average/mean. (18). Average/mean). (19). Space of seats in cm. (20). Height/altitude of passenger compartment m. (21). Maximum ... in t. (22). Maximum ... in t. (23). Maximum ... in %. (24). Maximum ... in t. (25). V (km/h km). (26). V kilometer per kilometer. (27). V km/h.

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REFERENCES.

1. M. F. Astakhov et al. Reference book according to the calculation of aircraft to the strength. M., the Oborongiz, 1954.

2. Aerodynamics of aircraft components at high speeds. Edited by A. F. Donoven and G. R. Lourena. M., IL, 1959.

3. A. A. Badyagin, Ye. A. Ovrutskiy. Design of passenger aircraft taking into account the economy of operation. M., "machine building", 1964.

4. V. F. Bolkhovitinov. Ways of the development of flight vehicles. M., "machine building", 1962.

5. U. Vudson and D. Konover. Handbook on human factors engineering for the engineers and artist-designers. M., "Mir", 1968, page 208.

6. A. L. Gimmel'far'. Working/treatment and the use of statistics

during the design of aircraft. The "transactions of the MAI", iss. 2, 1946. Designing calculation of the required reserve of fuel and gross weight of aircraft. The "transactions of the MAI", iss. 108, 1969.

7. V. V. Golubyov. Transactions on aerodynamics, M. 1., GITTL, 1957.

8. V. T. Goroshchenko. Aerodynamics of high-speed aircraft. M., Oborongiz, 1948, page 330.

9. S. M. Yeger. Design of passenger jet aircraft. M., "machine building", 1964.

10. B. M. Kagan, T. M. Ter-Mikaelyan. Solution of engineering problems in the digital computers. M., "Energy", 1964.

11. G. S. Kalachev. Indices of maneuverability, controllability and stability of aircraft. M., OBORONIGIZ, 1958.

12. S. N. Kahn, and A. I. Sverdlov. Calculation of aircraft to the strength. M., the Oborongiz, 1958.

13. P. M. Kreyson. Aircraft in 20 years ONTI, 1934. Weight characteristics of aircraft, 1935.

14. T. I. Ligum. Aerodynamics and the dynamics of the flight of turbojet aircraft. M., "transport", 1967.

15. A. K. Martynov. Experimental aerodynamics. M., Oborongiz, 1958.

16. Ye. A. Ovrutskiy. Economic evaluation of transport aircraft, RIO Aeroflot, 1940.

17. I. V. Ostoslavskiy. Aerodynamics of aircraft. M., Oborongiz, 1957.

18. I. V. Ostoslavskiy, I. V. Strazheva. Dynamics of flight. M., Oborongiz, 1963.

19. V. F. Pavlenko. VTOL aircraft. M., Voenizdat, 1966, page 144-186.

20. V. I. Polikovskiy, A. A. Badyagin. On the coefficient of an increase in the launching weight of flight vehicles, IVUZ, 1966, No. 1.

21. V. S. Pyshnov. Coefficients for the evaluation of aircraft. The "technique of the air fleet", 1945 No 5.

22. L. A. Rastrigin. The random search. Riga, "ZINATNE", 1965.

23. P. D. Samsonov. Design and the construction/design of seaplanes, ONTI NKTP, 1936.

24. F. I. Sklyanskiy. Control of supersonic aircraft. M., "Machine building", 1964, page 242-248.

25. D. L. Tomashevich. Construction/design and the economy of aircraft. M., Oborongiz, 1960.

26. N. N. Fadeyev. Theoretical formula of the weight of tapered wing. "transactions MAI", iss. 138, Oborongiz, 1961.

27. N. N. Fadeyev. Aviation weight. The "transactions of TsAGI [- Central Institute of Aerohydrodynamics im. N Ye Zhukovskiy]", iss. 470, 1940.

28-29. N. N. Fadeyev. Investigatiosn of the suitable sizes/dimensions of aircraft. The "transactions of TsAGI", iss. 461, 1939.

30. N. A. Fomin. Design of aircraft. M., Oborongiz, 1961.

31. V. B. Shavrov. History of the constructions of the aircraft in the USSR. M., "machine building", 1969.

32. V. M. Sheynin. Weight and transport effectiveness of passenger

aircraft, M., Oborongiz, 1962.

33. V. M. Sheynin. Position of center of gravity in aircraft. M., Oborongiz, 1952.

34. F. R. Shenli. Analysis of weight and strength of aircraft constructions/designs, Oborongiz, 1957.

35. L. E. Elsgolts. The calculus of variations. M. l., Gostekhteoretizdat [- State Publishing House of Technical and Theoretical Literature], 1952.

36. Everling E. Verglucks — grössen zur Flugzeugstatistick, ZFM, Nr. 10, 1926. Second International Conferens, No. 4, 1950, pp. 556—576.

37. The college of Aeronautics Cranfield Note, No. 77, Sept. 1957.

38. I. I. Shuneyko. Winged spacecraft, the AS USSR, 1966.

39. S. Yu. Skripnichenko. Variable sweepback. M., Voyenizdat, 1969.

40. S. Yu. Gaukhman, B. N. Kazanskiy. Bases of the development of hypersonic passenger aircraft. R., publishing house RKII GA, 1972.

41. Jane's all the World's Aircraft, London, 1968—1971.

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